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APOLLO 9 MISSION REPORT
SUPPLEMENT 5

SERVICE PROPULSION SYSTEM
FINAL FLIGHT EVALUATION

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SERVICE PROPULSION SYSTEM
FINAL FLIGHT EVALUATION

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SERVICE PROPULSION SYSTEM
FINAL FLIGHT EVALUATION

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1. PURPOSE AND SCOPE

The purpose of this report is to present the results of the postflight analysis of the Service Propulsion System (SPS) performance during the Apollo 9 Mission. The primary objective of the analysis was to determine the steady-state performance of the SPS under the environmental conditions of actual space flight. This report has been prepared as Supplement 5 to the Apollo 9 Mission Report (MSC-PA-R-69-2).

This report covers the additional analyses performed following the issuance of Reference 2.

The following items are the major additions to the results reported in Reference 2.

1) The steady-state performance as determined from analysis of the second and third SPS burns is presented.
2) The analysis techniques, problems and assumptions are discussed.
3) The flight analysis results are compared to the preflight predicted performance.
4) The pressurization system performance is discussed in greater depth.
5) The transient data and performance are included.
6) The SPS thermal control data are presented.

2. SUMMARY

The performance of the CSM 104 Service Propulsion System during the Apollo 9 Mission was evaluated and found to be satisfactory. The SPS performed eight maneuvers for a total firing time of 504.42 seconds.

The steady-state performance was determined by analyzing the second and third SPS burns using the Apollo Propulsion Analysis Program. The mixture ratio during both burns was less than predicted by 0.03 to 0.05, and was outside the 3 sigma limits associated with the flight prediction. The less-than-predicted mixture ratio resulted from the propellant tank ullage pressures being other than predicted, and from apparent errors in the engine hydraulic resistances. The fuel and oxidizer engine resistances, as determined from the flight analysis, were approximately 2.1% and 0.9% less, respectively, than the values determined from acceptance test data. The change required in the oxidizer resistance was within the expected tolerances of ± 1.9% (3-sigma) used in the preflight dispersion analysis (Reference 1). The fuel resistance change was approximately equal to the expected minus 3-sigma uncertainty of -2.1%. It was not possible to determine from the Apollo 9 flight data alone whether these apparent resistance errors result from the statistical uncertainties in the ground test data, systematic errors (biases) in the ground test data, or are indicative of some flight related phenomena not presently modeled, such as propellant helium absorption effects.

The differences between the predicted and actual tank pressures were believed to result from a change in helium regulator outlet pressure resulting from a parts replacement prior to launch, the use of specification values for the helium check valve pressure drops in the preflight simula-

tion, and possible errors in the helium line resistances used in the preflight simulations.

The engine performance corrected to standard inlet conditions for both the second and third burns was as follows: thrust 20728 pounds, specific impulse 313.9 seconds, and propellant mixture ratio 1.587. These values are 0.9% greater, 0.3% greater and 0.6% less, respectively, than the corresponding values determined from acceptance test data. The specific impulse and mixture ratio differences are within the expected tolerances; however, the standard inlet conditions thrust difference exceeded the 3-sigma limits of $\pm$ 0.73%.

The Propellant Utilization and Gaging System (PUGS) operation throughout the mission was satisfactory and as expected, with the exception of (1) the erroneous oxidizer storage tank reading caused by clinging propellant and drying of the dielectric compensator, and (2) the excessive propellant level stabilization times.

It was determined that prior to crossover the oxidizer sump tank level was above the top of the sump tank primary probe, resulting in approximately 100 lbm of oxidizer being ungageable until after crossover. This condition was attributed to the transfer of oxidizer from the storage tank to the sump tank because of helium absorption in the sump tank during the time from propellant loading to the first SPS burn.

The operation of the helium pressurization system, the SPS transient performance, and the SPS thermal characteristics were found to be satisfactory.

3. INTRODUCTION

The Apollo 9 Mission was the ninth in a series of flights using specification Apollo hardware. It included the second flight test and the first manned flight of the Lunar Module (LM), the third manned flight of Block II Command and Service Modules (CSM), and was the second manned flight using a Saturn V launch vehicle. The overall objectives of the mission were to evaluate crew operation of the Lunar Module and to demonstrate docked vehicle functions in an earth orbital mission, thereby qualifying the combined spacecraft for lunar flight. Combined spacecraft functions included Command Module docking with the Lunar Module, spacecraft ejection from the launch vehicle, five SPS firings while docked, a docked Descent Propulsion System (DPS) firing, and extravehicular crew operation from both the Lunar and Command Modules. Lunar Module operations included a complete rendezvous and docking profile and an Ascent Propulsion System (APS) firing to propellant depletion. Command and Service Module operations included, in addition to the five docked SPS firings, three undocked SPS firings.

The space vehicle was launched from the Kennedy Space Center (KSC) at 11:00:00 a.m. (EST) on 3 March 1969. Following a normal launch phase, the S-IVB stage inserted the spacecraft into an orbit of 102.3 by 103.9 nautical miles. The CSM docked with the LM and the docked spacecrafts were ejected from the S-IVB approximately four hours after launch.

There were eight SPS firings during the mission. The first SPS burn was a docked maneuver performed approximately 6 hours after liftoff which produced a velocity change of 36.6 ft/sec. At approximately 22 hours, the second docked SPS firing was performed, resulting in a velocity change of 850.5 ft/sec. The third, and longest, SPS burn was conducted at approximately 25 hours after liftoff. The third burn was a docked burn and produced a velocity change of 2567.9 ft/sec. Approximately 3 hours later, the fourth docked SPS burn was performed for a velocity change of 300.5 ft/sec. The fifth SPS firing, which was the final docked maneuver, was accomplished at approximately 54 hours after liftoff for a velocity change of 572.5 ft/sec. Prior to the fifth SPS burn, at approximately 50 hours, the first DPS burn was performed in the docked configuration for a duration of 371.5 seconds. At 93:39:36 (hr:min:sec) ground elapsed time (GET) the LM and CSM were separated in preparation for the rendezvous maneuvers. Between separation and the sixth SPS firing, at approximately 123 hours, the planned rendezvous maneuvers were successfully accomplished, including a DPS phasing burn of 19.7 seconds, a DPS insertion maneuver burn of 22.3 seconds, the descent-ascent stage separation, and an APS insertion maneuver burn of 15.6 seconds duration.

Following the ascent stage-CSM rendezvous, the ascent stage was jettisoned and the APS performed a burn to depletion of 362.4 seconds duration. The sixth SPS burn produced a velocity change of 33.7 ft/sec. Approximately 2 days later, the SPS was activated for the seventh firing. The seventh SPS burn resulted in a 650.1 ft/sec velocity change, and was designed to give a burn time approximately 15 seconds longer than the preflight planned duration of 10 seconds, in order to accomplish a SPS gaging system test that had been added to the firing objectives. The eighth, and final, SPS firing was the de-orbit burn and occurred approximately 240 hours after liftoff. The resultant velocity change was 322.7 ft/sec.

The actual ignition times and burn durations for the eight SPS firings are shown in Table 1.

The Apollo 9 Mission utilized CSM 104 which was equipped with SPS Engine S/N 62 (Injector S/N 120). The engine configuration and expected performance characteristics (Reference 1) are contained in Table 2.

The Apollo 9 SPS configuration was very similar to the Apollo 7 and 8 configurations, which were the two previous flight Block II Apollo spacecrafts.

The SPS engine was started in the single bore engine valve mode on all eight burns to reduce the magnitude of the chamber pressure overshoot experienced on previous flights when starting in the dual bore mode. During the second, third, fourth, fifth, seventh, and eighth burns the other bore was opened 3 to 4 seconds after ignition and the remainder of the burn was performed in the dual bore mode. The SPS PU valve was left in the normal position throughout the mission.

The first three SPS maneuvers were no-ullage starts, while the fourth and fifth maneuvers were preceded by 20 second, 4-jet, +X SM RCS ullage maneuvers to insure SPS propellant settling. The sixth, seventh and eighth burns were each preceded by 20 second, 2-jet +X SM RCS ullage maneuvers.

There was no Apollo 9 Mission Detailed Test Objective specifically related to the SPS.

4. REVIEW OF SPS OPERATION

The SPS telemetry (T/M) data recorded during boost and the 6-hour coast period prior to the first SPS burn indicated that all SPS pressures and temperatures had remained within their normal operation ranges during that time period, verifying that the launch and spacecraft separation from the S-IVB had no adverse effects on the SPS.

The SPS operation was normal and satisfactory during each of the eight firings accomplished during the Apollo 9 Mission, with the exception of the operation of the propellant utilization and gaging system (PUGS), which was not as expected. The PUGS operation is discussed in detail in the PUGS Evaluation Section. The available T/M data indicated that all SPS temperatures and pressures remained within the ranges of nominal operation during all burn and coast phases of the mission. Table 3 contains representative values of the steady-state pressures measured during each of the eight firings. These data indicate that the SPS operation was consistent throughout the mission.

The fifth SPS maneuver followed a docked DPS burn of approximately 367 seconds. Preflight analyses indicated that when the CSM and LM are docked, and a DPS burn is performed, a negative acceleration that exceeds approximately 0.1 foot/second^2 will, in time, cause depletion of the SPS propellants captured under the retaining screens. However, the retention reservoir should remain full. The ullage maneuver conducted in preparation for a subsequent SPS firing will bring the propellant down to the top screen with a portion of the propellant passing through the screens until the entire screen area is wet, which prevents displaced helium (below the screen) from passing forward. An SPS firing under this condition causes the propellant to drop through the two screens rapidly to displace the gas but not before some gas is brought into the propellant retention reservoir. Analysis indicated that a large portion of the quantity of gas entering the reservoir may be captured without an excessive rate of gas ingestion into the engine. Test data indicate that the gas in the reservoir will be slowly entrained with the propellant flow into the engine, and the reservoir should be restored from gaseous to normal operation after about 40 seconds of continuous SPS operation. The performance during the fifth SPS maneuver was normal and smooth. The subsequent sixth SPS burn was also smooth, verifying that the 41-second fifth burn was of sufficient duration to remove the trapped helium and restore the reservoir to normal operation (liquid) and thereby prevent any subsequent accumulation of gas bubbles near the engine valve inlets.

The sixth SPS firing was originally scheduled for the 77th revolution and was to be controlled by the digital autopilot (DAP). The planned DAP configuration for the sixth burn included an automatic +X, 2-jet, 20 second SM RCS ullage maneuver prior to SPS ignition. The +X translation was not achieved at the proper time and the SPS firing was aborted. The failure of the +X translation maneuver to occur resulted from improper programming of the DAP configuration changes prior to the planned SPS burn. The SPS firing was rescheduled for the 78th revolution and was successfully accomplished at 123:25:07 GET.

Throughout the coast prior to the first SPS burn, the measured fuel tank pressure (SP0006P) consistently read about 2 psi less than the measured fuel interface pressure (SP0930P). During zero-g coast the two measurements should indicate equal pressures. Comparisons of data from the two measurements during other coast periods also showed a similar disagreement. Data recorded several hours prior to launch also indicated a 2 psi discrepancy between the fuel tank and interface measured pressures, after accounting for head effects. Prior to launch, the fuel tank pressure read about 2-3 psi less than the oxidizer tank pressure. The oxidizer tank and interface pressures agreed well prior to launch and during the coast to the first SPS burn. Based on these comparisons, it was concluded that the fuel tank pressure was most probably biased approximately -2 psi throughout the mission.

More detailed discussions of the operation of the helium pressurization system, the thermal control system and the SPS transient performance are contained in their respective sections of this report.

5. STEADY-STATE PERFORMANCE ANALYSIS

Analysis Technique

The major analysis effort for this report was concentrated on determining the flight steady-state performance of the SPS during the second and third burns. The remaining six burns were of insufficient duration to warrant detailed performance analysis. The performance analysis was accomplished with the aid of the Apollo Propulsion Analysis Program which utilizes a minimum variance technique to "best" correlate the available flight and ground test data. The program embodies error models for the various flight and ground test data that are used as inputs, and by statistical and iterative methods arrives at estimations of the system performance history, propellant weights and spacecraft weight which "best" (minimum-variance sense) reconcile the available data.

Analysis Description

The steady-state performance analysis utilized data from the flight measurements listed in Table 4. Early in the postflight analysis it became apparent that the flight performance, as indicated by the T/M data, was not totally consistent with the system model as established from ground test data. The inflight mixture ratio, based on the PUGS data, was less than expected, and the deviation could not be completely explained by differences in the measured system pressures or propellant conditions. Initial simulations also indicated that the average thrust for the two burns analyzed was significantly greater than expected for the measured system pressures and propellant conditions, when using the preflight model. Because of these apparent performance inconsistencies, it was necessary to deviate somewhat from the analysis approach used in previous SPS evaluations. In addition to the performance inconsistencies, several data and modeling problems were encountered which required that certain assumptions be made before meaningful simulations of the two burns in question could be accomplished.

The analysis procedures used, along with the data and modeling problems encountered, and the assumptions which were required to resolve them are described in detail in the following paragraphs.

As previously discussed, the inflight mixture ratio and thrust were not consistent with the preflight system model as established from ground test data. From preliminary simulations it was determined that in order to achieve an acceptable match to the flight data, it would be necessary to allow the analysis program to adjust the engine hydraulic resistances, i.e. consider them as state variables. In previous SPS postflight analyses the engine resistances have been assumed known from ground test data, and were input to the program as constants.

It was further observed (see PUGS Evaluation) that the oxidizer sump tank level prior to propellant crossover (storage tank depletion) was above the maximum gageable height because of oxidizer transfer between the tanks. This meant that accurate oxidizer gaging data were not available during the second burn, and the initial portion of the third burn. The PUGS data also indicated that some fuel transfer may have also occurred between the storage and sump tanks, although the sump tank level did remain below the maximum gageable height. Because of these factors, it was necessary to consider not only the total oxidizer and total fuel masses as state variables, but to also consider their respective amounts in each tank as unknowns prior to crossover.

The consideration of the additional variables, or "degrees of freedom," complicates the analysis, especially for those portions of the SPS duty-cycle prior to crossover. Therefore, it was decided to first analyze the portion of the third burn following crossover, in order to determine the engine hydraulic resistances, and to then use those resistances as constants in analyzing the second burn. This was an iterative procedure in which the SPS performance was determined primarily from the third burn analysis, and then verified by the simulation of second burn. The steady-state performance during the third burn was derived from the analysis of a 164-second segment of the burn. The segment analyzed began approximately 97-seconds following ignition (FS-1), excluding that portion of burn prior to crossover, and included the flight time between 91156 and 91320 seconds G.E.T. The steady-state performance during the second burn was determined from the analysis of a 85-second segment of the burn. The segment of the burn analyzed commenced approximately 21 seconds after SPS ignition (FS-1), and included the flight time between 79945 and 80030 seconds G.E.T. The first 21 seconds of the burn were not included, in order to minimize any errors resulting from data filtering spans which include transient data, and because the PUGS data near the start of the burn was erroneous (see PUGS EVALUATION AND PROPELLANT LOADING). The time segment analyzed was terminated approximately 4 seconds prior to SPS shutdown (FS-2) for similar reasons.

The ability to separate errors in the computed thrust from errors in the estimated initial spacecraft mass was reduced when compared to previous SPS analyses because of the increased spacecraft mass associated with the docked configuration. This condition exists because the increased mass decreases the rate of change of vehicle acceleration, which is the primary parameter in separating thrust and mass errors. Both the second and third burns were conducted in the docked configuration. Therefore, it was necessary to assume that the vehicle damp weight (CSM/LM minus SPS propellants) was known for each burn analyzed. The estimated spacecraft damp weight at ignition for both burns was obtained from the Apollo Spacecraft Program Office, and was assumed constant throughout the burn. The damp weight used for the second burn was 54760 lbm, and the weight used for the third burn was 54750 lbm. The initial estimates of the SPS propellants onboard at the beginning of the time segment analyzed for the second burn were extrapolated from the loaded propellant weights. The initial propellant estimates for the the time segment analyzed for the third burn were extrapolated from the computed propellants remaining at the end of the time segment analyzed for the second burn. In all cases the extrapolations of propellant masses used to establish the initial estimates for a given simulation were performed in an iterative manner and utilized the derived flowrates from the preceding simulation. This insured that the derived propellant mass history was consistent between the two time segments analyzed.

As previously discussed (see Review of SPS Operation), the measured fuel tank pressure data (SP0006P) was adjudged to be biased by approximately -2 psi. Therefore, a -2 psi bias was input to the analysis program as an initial estimate of the mean error for these data.

The transducers for both the oxidizer and fuel tank pressure flight measurements are not located in the storage tank ullage, but are in the helium lines to the tanks; between the heat exchangers and the tanks. The theoretical pressure drops (friction losses) for the oxidizer and fuel helium lines between the measurement locations and the storage tank ullages were computed to be approximately 0.6 psi and 0.5 psi, respectively, at the nominal helium flowrates. The equations to compute these pressure drops as functions of the helium flowrates were incorporated in the analysis program measurement model. These pressure drops had been assumed to be zero in previous flight analyses, and their inclusion improves the measurement model.

The SPS engine thrust chamber throat area was input to the program as a function of the time from ignition for each burn. The throat area time history, shown in Figure 1 , was assumed to be the same as predicted (Reference 1).

The SPS propellant densities used in the analysis were calculated from propellant sample specific gravity data obtained from KSC, flight propellant temperature data, and flight interface pressures. The temperatures used were based on data from all the feed-system and engine feedline temperature measurements and were input to the program as functions of time. During steady-state operation, it was assumed that for both oxidizer and fuel, their respective tank bulk temperatures and engine interface temperatures were equal.

Both the second and third burn simulations were performed using an "ullage pressure driven" SPS model. Simply stated, this model utilizes input oxidizer and fuel storage tank ullage pressure values, as functions of time, for the starting points in computing the pressures and flowrates throughout the system. The input tank pressures used are generally the filtered data from the flight tank pressure measurements. The program is free to bias the input pressures, if so required to achieve a minimum variance solution, but the version used (Linear Model 0) is essentially constrained to follow the shape of the input tank pressure profiles. The shape of the tank pressure profiles, in turn, strongly influence the computed thrust shape, and therefore, the calculated acceleration shape. The initial simulations of the third burn, using the filtered tank pressure data, yielded computed acceleration shape errors. The errors were seen to be directly correlated with the "steps" in the filtered tank pressure data [(1)], which are shown in Figure 2. The steps in question occur because of the PCM quantization of the data, which for the tank pressure data is approximately 1 psi/PCM count, and are not considered to best reflect the actual tank pressure profiles. By utilizing the noise-in-the-state version (Linear Model 2) of the program it was possible to derive tank pressure profiles which are considered more realistic. The derived profiles, which are also shown in Figure 2, were then input to the Linear Model 0 version of the program for subsequent simulations.

Analysis Results

The resulting values of the more significant SPS performance parameters, as calculated in the analysis program simulation of the second burn, are contained in Table 5, along with their corresponding preflight predicted values. The values presented are for a time slice 90 seconds after FS-1, and, although the SPS performance is time dependent, these values are considered representative of the performance throughout the entire segment of the second burn that was analyzed. Table 6 presents the calculated performance values from the third burn simulation, along with the predicted values, for a time slice 225 seconds following FS-1. Again, these values are considered representative of the performance throughout the burn segment that was simulated.

[(1)] The filtered data shown in Figure 2 were adjusted to account for the pressure drops between the measurement location and the tanks, and for the assumed bias in the fuel tank data previously discussed in order to be comparable to the predicted and derived storage tank pressures.

Figure 3 shows the calculated SPS specific impulse, propellant mixture ratio, and thrust, as functions of time, for the two burn segments analyzed. For comparison, Figure 3 also contains the predicted performance for the entire second and third burns. As shown, the specific impulse was essentially a constant value of 313.9 seconds throughout the burn segments analyzed. Based on the values computed for the two burn segments analyzed, and the qualitative comparison of the data from all eight burns (Review of SPS Operation), it is concluded that the SPS steady-state performance throughout the entire mission was satisfactory. The propellant mixture ratio, however, was significantly less (by 0.03 to 0.05) than predicted during both burns. A detailed comparison of the flight performance to the predicted performance is contained in a following section.

The program determined that the best match to the available data required that the engine hydraulic resistances be adjusted from their preflight values. The derived fuel resistance was 854.4 lbf-sec^2/lbm-ft^5, and the derived oxidizer resistance was 484.1 lbf-sec^2/lbm-ft^5. These values are approximately 2.1% and 0.9% less, respectively, than the values determined from engine acceptance test data and used in the preflight prediction.

As observed in previous SPS flight analyses, the measured chamber pressure exhibited an apparent positive drift during both the second and third burns, when compared to the program calculated chamber pressure. The average magnitudes of the drift, over the segments of the burns analyzed, were approximately 0.016 psi/sec and 0.003 psi/sec for the second and third burns, respectively. The drift is believed to result from thermal effects on the transducer. The drift rate has been observed to be most pronounced following ignition and to decrease with time from ignition. This characteristic seems to support the hypothesis that the drift is thermally induced since the injector flange temperature (measurements SP0061T and SP0062T) exhibited similar trends, i.e., a high rise rate immediately following ignition, with the rate of rise then decreasing with time from ignition. Since the burn segment simulated for the third burn commenced 97 seconds after ignition, compared to 21 seconds for the second burn segment, the smaller average drift rate observed for the third burn segment is as expected.

During both burns, the measured oxidizer and fuel interface pressure data (SP0931P and SP0930P) appeared biased by approximately -2 psi and -1 psi, respectively. Although the apparent biases are well within the instrumentation uncertainties, recent discussions with North American Rockwell (NR) have revealed that there is some concern that the flight interface measurements may either be systematically biased, or are not sensing pressures consistent with the theoretical interface locations on which the original Block II SPS feed-system hydraulic resistances were based. This concern apparently arose primarily from NR's difficulties in correlating flight data from previous SPS missions, and, at present, is unresolved. Analysis of this question should be continued, and it is recommended that the applicable data from the White Sands Test Facility SPS testing be reviewed since significant instrumentation redundancy exists there that is not available in flight.

The analysis indicated that the preflight chamber throat area (Figure 1) which was used in the postflight simulations, was relatively accurate, in that no changes were required to achieve a satisfactory data match for either the second or third burn. This conclusion is somewhat counter to the Apollo 8 analysis results (Reference 3), which required some adjustments to the predicted throat area in order to match the flight data for the fourth burn, which was the second long duration burn on that mission. There were differences, however, in the Apollo 8 and Apollo 9 SPS duty-cycles (burn durations and time between burns) which might explain the apparent differences in the throat area characteristics. The SPS throat area characteristics during future flights should be investigated in order to validate the characterization being used for the preflight predictions.

The computed oxidizer and fuel masses at the start of the second burn simulation were 104 lbm and 95 lbm less, respectively, than the initial estimates as extrapolated from the reported propellant loads (see PUGS EVALUATION AND PROPELLANT LOADING). These differences are not considered significant and may partially reflect errors in the vehicle damp weight and in the extrapolation, as opposed to errors in the loading data. Based on the analysis of the second burn, it was determined that approximately 100 lbm of oxidizer was ungageable prior to crossover because of the oxidizer transfer between tanks (see PUGS EVALUATION AND PROPELLANT LOADING).

Critique of Analysis

Shown in Figures 4 through 21 are analysis program output plots which represent the residual errors, or differences between the filtered flight data and the program-calculated values. The figures represent vehicle thrust acceleration, oxidizer tank pressure, fuel tank pressure, fuel interface pressure, oxidizer interface pressure, oxidizer sump tank quantity, fuel sump tank quantity, chamber pressure, oxidizer storage tank quantity, and fuel storage tank quantity, in that order. The first set of residual plots is for the second burn analysis, and the second set is for the third burn analysis. No storage tank quantity plots are shown for the third burn since the storage tanks are empty after crossover. The filtered flight data is also included on each plot.

A strong indication of the validity of the analysis program simulation can be obtained by comparing the thrust acceleration calculated in the simulation to that derived from the Apollo Command Module Computer (CMC) ΔV data transmitted via measurement CG0001V. Figures 4 and 14 show the thrust acceleration during the portion of the burns analyzed, as derived from the CMC data, and the residual error between the CMC and program calculated values. The residual error time histories have essentially zero means and little, if any, discernible trends. This indicates the simulations are relatively valid, although other factors must also be considered in critiquing them.

As previously discussed, the measured chamber pressure data was observed to drift with burn time. Although this drift was partially modeled from previous SPS flight analysis results, it still significantly compromised the usefulness of this measurement as far as detailed performance analysis was concerned. Furthermore, on this flight it was observed that the measured chamber pressure prior to, and following, the eight SPS burns indicated values ranging between -5 psi and +1.5 psi, when it should have indicated zero pressure. Because of these inconsistent "zero shifts", and the drift with time, this measurement was considered essentially useless for the detailed analysis, where chamber pressure differences of less than 1 psi are significant, and was therefore not used in the simulations. The residual errors plots, Figures 11 and 21, for the chamber pressure during the second and third burns are included for information only. Because the measured chamber pressure could not be utilized, the program's ability to distinguish tank and interface pressure measurement errors from errors in the preflight engine model (engine resistances, chamber characteristic velocity, and specific impulse) was somewhat diminished.

In the simulations of both burns it was assumed that the measured fuel tank pressure data was biased by -2 psi. This assumption was based on the differences between the measured fuel tank and fuel interface pressures during coast (see Review of SPS Operation), and the analysis results appear to support this assumption. However, it is conceivable that the measured fuel tank pressure was correct and that the measured fuel interface pressure was biased +2 psi. Because the assumption concerning the fuel tank/fuel interface pressure biases was relatively significant, the effects of making the latter assumption were investigated for the third burn. A simulation of the third burn was made assuming the measured fuel tank pressure to be unbiased and assuming the measured fuel interface pressure to be biased +2 psi. The calculated flight performance from the simulation was not significantly different than shown in Table 6. The thrust was the same and the specific impulse and mixture ratio were 0.4 seconds less and 0.01 greater, respectively, than the reported simulation. However, the simulation in question required even larger adjustments (-4% for fuel and -2.27% for oxidizer) to the engine resistances than the reported simulation. It is concluded, therefore, that the assumption about the fuel tank pressure was reasonable, and even if wrong would not greatly alter the conclusions contained in this report.

Comparison with Preflight Performance Prediction

Prior to the Apollo 9 Mission the expected performance of the SPS was presented in Reference 1. This performance prediction was for the integrated propellant feed/engine system and, wherever possible, utilized data and characteristics for the specific SPS hardware on this flight.

The predicted steady-state thrust, specific impulse, and propellant mixture ratio for the second and third burns are shown in Figure 3 versus the time from ignition for each burn. Also shown, for comparison, are the corresponding analysis program calculated flight performance histories for the portions of the two burns which were analyzed. As shown in Figure 3, and previously in Tables 5 and 6, the computed thrust and specific impulse, were within the expected tolerances throughout the burn segments analyzed.

The computed propellant mixture ratio throughout both burn segments analyzed is seen in Figure 3 to be significantly less (as much as 0.05) than predicted, and well outside the -3 sigma limits presented in Reference 1. The less-than-predicted mixture ratio resulted, in part, from the propellant tank ullage pressures being different than predicted. As shown in Figure 2, throughout the burn segments analyzed, the oxidizer tank pressure was generally less than predicted by 1 to 3 psi, and the fuel tank pressure was greater than predicted by 0.7 to 2.0 psi. Based on a linearized engine model (influence coefficients) a 1 psi reduction in oxidizer tank pressure combined with a 1 psi increase in fuel tank pressure would, for example, result in a reduction in mixture ratio of approximately .02.

The predicted tank pressures in Reference 1 were based, in part, on the CSM 104 helium regulator acceptance test data. However, prior to flight the helium regulator controller section stems were replaced because of quality faults found in similar stems. Variations in manufacturing tolerances of these stems are believed to be such that the regulated pressure with the new stems could have been significantly different than that during the acceptance tests. Also, at the time that the analysis in Reference 1 was performed, the helium check valve acceptance test ΔP (pressure drop) data for the check valves on CSM 104 were not available, and specification values were used. These factors partially account for the differences between the predicted and flight tank pressures.

Past Block II SPS predictions, including Apollo 9, have generally shown the expected steady-state oxidizer tank pressure to be about 2 psi greater than the expected fuel tank pressure. The reason for this predicted difference between the oxidizer and fuel tank pressure is that the present helium line resistances between the regulator and the tanks are significantly different for the oxidizer and fuel side. The present values for the helium line resistances were furnished by NR in Reference 4, and are believed to have been determined theoretically. The computed tank pressures (Figure 2) for the second and third burns showed oxidizer pressure less than fuel. The Apollo 8 analysis (Reference 3) showed the oxidizer tank pressure to be within 1 psi of the fuel tank pressure. It is, therefore, recommended that the helium line resistances used for future SPS predictions be re-evaluated to more accurately predict flight tank pressures. It is further recommended that the uncertainties in the helium line resistances and check valve ΔP's be included in the SPS preflight dispersions analysis. Uncertainties in helium regulator outlet pressure were the only dispersions considered in Reference 1 that significantly affect tank pressures. Furthermore, errors in regulator outlet pressure essentially change oxidizer and fuel tank pressures the same amount, and therefore, have negligable effect on mixture ratio. The inclusion of the uncertainties in helium line resistances and check valve ΔP's, which affect oxidizer and fuel tank pressures independently, should result in more realistic mixture ratio dispersions.

Although the engine thrust was within the expected tolerances, it was greater than expected for the flight tank pressures and propellant temperatures, as evidenced by greater than predicted standard inlet conditions thrust (see Engine Performance at Standard Inlet Conditions). In order to account for the unexplained portion of the decreased mixture ratio and the increased thrust, the engine resistances were adjusted, with the fuel resistances decreasing approximately 2.1% and the oxidizer resistance decreasing about 0.9%. The change required in the oxidizer resistance was within the expected tolerances of $\pm$ 1.9% (3-sigma) used in the preflight dispersion analysis (Reference 1). The fuel resistance change was approximately equal to the expected minus 3-sigma uncertainty of -2.1%. It is not possible to determine from the Apollo 9 flight data alone whether these apparent resistance errors result from the statistical uncertainties in the ground test data, systematic errors (biases) in the ground test data, or are indicative of some flight related phenomena not presently modeled, such as propellant helium absorption effects. Therefore, it is recommended that the Apollo 10 postflight analysis concentrate on this problem to determine whether the apparent resistance errors occur consistently on SPS flights. It is further recommended that the SPS engine acceptance test instrumentation and procedures be examined to determine whether a systematic ground test error(s), such as a flowmeter bias, could exist. Although to date no definite propellant helium absorption performance effects have been identified for the SPS engine, it is recommended that any available information and/or test data pertaining to helium absorption effects on the SPS, or similar, engines be evaluated in relation to the flight results reported herein.

Engine Performance at Standard Inlet Conditions

The expected flight performance of the SPS engine was based on data obtained during the engine and injector acceptance tests. In order to provide a common basis for comparing engine performance, the acceptance test performance is adjusted to standard inlet conditions. This allows actual engine performance variations to be separated from performance variations which are induced by feed-system, pressurization system, and propellant temperature variations.

Engine flight performance, as determined in the steady-state analysis, and corrected to standard inlet conditions, yielded essentially identical results for both the second and third burns. The standard inlet conditions thrust, specific impulse, and propellant mixture ratio were 20728 pounds, 313.9 seconds, and 1.587, respectively. These values are 0.9% greater, 0.3% greater, and 0.6% less, respectively, than the corresponding values computed from the engine model used in the preflight prediction. The preflight engine model was established from acceptance test data, and the standard inlet conditions performance values computed with it agree well with the values reported in the engine acceptance test log.

The flight standard inlet conditions thrust was slightly greater than the upper acceptance test specification limit of 20705 pounds, and the 0.9% difference from predicted was greater than the expected tolerances of $\pm$ 0.73% (3-sigma). The mixture ratio difference was within the expected tolerances. The standard inlet conditions performance values reported herein were calculated for the following conditions:

STANDARD INLET CONDITIONS

Oxidizer interface pressure, psia	162
Fuel interface pressure, psia	169
Oxidizer interface temperature, °F	70
Fuel interface temperature, °F	70
Oxidizer density, lbm/ft^3	90.15
Fuel density, lbm/ft^3	56.31
Thrust acceleration, lbf/lbm	1.0
Throat area (initial value), in^2	121.602

Of primary concern in the flight analysis of all Block II engines will be the verification of the present methods of extrapolating the specific impulse for the actual flight environment from data obtained during ground acceptance tests at sea level conditions. Since the SPS engine is not altitude tested during the acceptance tests, the expected specific impulse is calculated from the data obtained in the injector sea level acceptance tests using conversion factors determined from AEDC qualification testing. As previously discussed, the standard inlet conditions specific impulse determined from analyses of the second and third burns was 313.9 seconds. The predicted specific impulse at standard inlet conditions, as extrapolated from the ground test data was 313.0 seconds. The expected tolerances associated with this predicted value (Reference 1) were ± 1.59 seconds (3-sigma). The flight value for both burns was well within these tolerances. Therefore, it is concluded that the present methods of extrapolating the expected flight specific impulse from the ground test data were satisfactory for this flight, and there is no evidence to warrant changing the methods for future flights. The validity of this conclusion should be continually verified on each subsequent flight.

6. PUGS EVALUATION AND PROPELLANT LOADING

Propellant Loading

The PUGS operated normally during SPS propellant loading. The oxidizer tanks were loaded to a CM display readout of 89.15% at a tank pressure of 110 psia and an oxidizer temperature of 70°F. The fuel tanks were loaded at 111 psia and 70°F to a display readout of 89.25%. The SPS propellant loads calculated from these data, and propellant sample density data, are shown in Table 7 in the 110 psia column. When the tank pressures were increased to 177 psia for oxidizer and 175 psia for fuel during the leak test, the CM displays read 88.83% for oxidizer and 88.65% for fuel. A decrease in the readings is expected when the tank pressures are increased because of tank stretch; however, the decreases observed were slightly greater than expected. The propellant masses computed from these readings are included in Table 7 under the 175 psia column. The oxidizer and fuel masses computed from the readings taken at the flight pressure were 21 lbm and 57 lbm less, respectively, than those computed from the readings taken at the loading pressure. There is no explanation for the differences in the masses computed at the two pressures, although it is possible that the tank stretch effects presently assumed are somewhat in error.

During ground checkout fuel point sensors #7, #8, and #15 gave failed indications.

PUGS Operation In Flight

The PUGS mode selection switch was set in the "normal" position for the first SPS maneuver, and the PU valve was in the "normal" position. Following the 4.5-second lockout period after ignition, the fuel storage tank T/M data indicated a rapid level increase. The oxidizer storage tank T/M data also indicated a level increase. These data indicate that the propellant levels inside the gaging stillwells were not stabilized by 4.5-seconds after ignition. Because of the low acceleration level (0.2 g) resulting from the large spacecraft mass, and the fact that no RCS ullage maneuver was performed, it is believed that the capillary and viscous effects inside the gaging system stillwells were more significant than experienced on previous flights, or during ground testing, and therefore the levels required longer to stabilize. Because of the low PUGS T/M data sample rate (1 sample/second) it is not possible to determine the amplitude or frequency of the level oscillations. The first burn cutoff occurred 5.2 seconds after ignition. The crew reported an oxidizer display quantity of 89.2%, a fuel quantity of 93.7%, and that the unbalance meter was pegged in the "decrease" position following the burn. The display readings would give an unbalance of about 1100 pounds, "decrease," which would peg the unbalance meter, which reads between 600 lbm "increase" and 600 lbm "decrease." No SPS PU Sensor Caution and Warning (C&W) light activation occurred during the first burn because the power to comparator units is delayed for approximately 5.5 seconds following ignition.

The second SPS maneuver was performed with the PUGS in the "normal" mode. The crew reported that shortly after ignition the SPS PU Sensor C&W light activated and that the unbalance meter was cycling over a large range, At the end of the burn the crew reported a displayed oxidizer quantity of 69.25%, a fuel quantity of 69.4%, and an unbalance of 30 lbm, "decrease." As previously discussed the level oscillations during the first burn caused the indicated quantities at ignition of the second burn to be approximately 89.2% oxidizer and 93.7% fuel, and the unbalance meter to be pegged in the "decrease" position. It has been computed that the actual quantities were approximately 87.9% oxidizer and 87.7% fuel. Therefore, when the lockout period following second burn ignition ended, the indicated quantities, especially the fuel, were significantly different from the actual quantities. The T/M data showed that following the lockout period the storage tank stillwell levels did not stabilize until 20 to 25 seconds after ignition. These initial level fluctuations are attributed to the low-g/capillary and viscous effects discussed previously, and were the cause of the unbalance meter cycling and the C&W light activation.

The T/M data showed that after about 25 seconds of burn the storage tank levels were decreasing at approximately a normal rate. At the end of the burn the T/M data gave an oxidizer quantity of 69.2% and a fuel quantity of 69.1%. These values agree essentially with the crew reported CM display values. The PUGS operation during the second burn was normal. The possibility of a high unbalance indication near ignition of the second burn, because of the effects noted during the first burn, was recognized and the crew was informed of the possibility prior to the burn.

The third SPS maneuver commenced with the PUGS in the "normal" mode. The crew reported that the SPS PU Sensor C&W light was activated approximately 6 seconds following ignition. Approximately 90 seconds later, after crossover (storage tank depletion) had occurred, the unbalance meter went to almost full scale on the "increase" side and the C&W light again activated. In the following 25 seconds, the C&W light activated four more times and the unbalance meter continued to show readings of over 500 lbm, "increase." At approximately 123 seconds following ignition the crew moved the PUGS mode selection switch to the "auxiliary" position. During the next 65 seconds the unbalance meter reading was reported to be fluctuating, and another C&W light was activated. The crew returned the PUGS mode selection swtich to the "normal" position about 188 seconds after ignition and it remained in that position throughout the remainder of the burn. Another C&W light activation occurred at 190 seconds after ignition. Following cutoff the crew reported the CM display readouts as 23.1% for oxidizer, 21.1% for fuel, and an unbalance meter reading of 500 lbm, "increase."

Figures 22 and 23 show the T/M PUGS data for the third burn. Figure 24 shows the indicated unbalance at selected times, as calculated from the T/M data. The indicated unbalance history shown in Figure 24 should reflect the CM display unbalance history, within the T/M accuracy. Also shown in Figure 24 are the times at which C&W light activations were recorded. The SPS PU Sensor C&W light will activate whenever the unbalance exceeds the critical propellant unbalance limits. The nominal critical unbalance limits, which are functions of the oxidizer quantity remaining, are also shown in Figure 24.

As shown in Figure 24, the first C&W light activation occurred at 6 seconds after ignition when an unbalance of greater than 700 lbm "decrease," was indicated. Figures 22 and 23 show that this unbalance transient was caused by a high reading on the fuel storage tank primary probe. The high reading on the fuel probe is attributed to the propellant levels not being stabilized when the lockout period ended. After about 25 seconds following ignition the levels had stabilized and Figures 22 and 23 show normal storage tank depletion rates. Although the unbalance was increasing somewhat prior to crossover, Figure 24 shows no excessive unbalance after stabilization, until following crossover. Following crossover, the indicated unbalance is seen to increase rapidly, exceeding the "increase" critical propellant unbalance limit, thereby initiating a C&W light activation approximately 95 seconds after ignition. The indicated unbalance apparently fluctuated about the "increase" critical limit initiating four more C&W light activations.

When the PUGS is in either the normal or primary mode, the displayed oxidizer and fuel total quantities are obtained by summing the storage and sump tank primary probe readings. The rapid increase seen in the unbalance after crossover was caused primarily by the oxidizer storage tank reading not being zero after crossover as expected. Figure 24 shows that the oxidizer storage tank reading decayed to about 0.2% at crossover, then increased to approximately 1.9% during the next 15 seconds, then decreased slightly to about 1.7% before the PUGS was switched to auxiliary. The fuel storage tank probe reading was zero after crossover, as expected.

The 0.2% oxidizer storage tank probe reading at depletion is attributed to the fact that the probe is zero point calibrated at KSC with no oxidizer in the tank; i.e., the probe is calibrated "dry." It is believed that during flight some residual oxidizer clings to the probe after storage tank depletion thereby giving a positive reading. This condition has been observed on previous flights, although the erroneous indications were less than observed on this flight. It is felt that possibly the low-g condition increases the amount of propellant that clings to the probe thereby accounting for the higher value on this flight.

The increase of this 0.2% reading to 1.9% is attributed to the characteristics of the dielectric compensator located near the bottom of the probe. The compensator, which is in a feed-back loop in the PUGS circuit, is covered by oxidizer (wet) in normal operation. Circuit analysis by the PUGS manufacturer has revealed that if the compensator is not completely covered by oxidizer, the probe output may be magnified by a factor of approximately 10:1. It is felt that helium flowing by the compensator following crossover eventually "dried" it enough to cause the observed increase in the probe reading from 0.2% to 1.9%. The 1.9% oxidizer storage tank probe reading accounts for approximately 460 lbm of the indicated unbalance, which when added to a sump tank unbalance of approximately 200 lbm caused the total indicated unbalance to exceed the "increase" critical propellant unbalance limit and activate the C&W light five times prior to the time the PUGS was switched to auxiliary.

A second factor which also contributed to the step increase in the unbalance after crossover was that the oxidizer level in the sump tank was apparently over the top of sensing element of the sump tank primary probe. During the first two burns and third burn prior to crossover, the oxidizer sump tank gage T/M data read 57.2% even though the maximum gageable is 57%. This meant that some of the oxidizer in the sump tank was ungageable prior to crossover and would explain why the 200 lbm sump tank unbalance was not sensed prior to crossover, i.e. prior to crossover the unbalance was less than 200 lbm increase. It is estimated that approximately 100 lbm of oxidizer was above the top of the sump tank probe prior to crossover. It is believed that helium absorbed from the sump tank ullage into the oxidizer liquid from the time of loading resulted in the transfer of oxidizer from the storage tank to the sump tank in order to reduce the sump tank ullage volume by the amount necessary to maintain a pressure balance between the tanks. Analytical calculations verify that this mechanism of transfer is feasible and could result in a final oxidizer level which is above the maximum gageable in the sump tank.

When the PUGS was switched to the auxiliary mode, the indicated unbalance immediately changed to approximately 50 lb "decrease." When in the auxiliary mode, the total oxidizer and fuel quantities for the CM displays are taken from the auxiliary (point sensor) gaging system. Figures 22 and 23 show the auxiliary gaging system T/M data. The step changes in the auxiliary oxidizer and fuel readings at approximately 136 and 143 seconds after ignition, respectively, are the resets caused by uncovery of fuel point sensor #9 and oxidizer point sensor #9, respectively. The inordinately large change (approximately 2.5%) in the fuel auxiliary reading at the uncovery of point sensor #9 is attributed to point sensors #7 and #8 failing to initiate resets, and to the fact that the preset integration rate between point sensors is approximately 10% lower than the actual flowrate. During loading, both fuel point sensor #7 and #8 failed to give proper indications on the ACE (ground displays). However, the exact failure mode could not be determined. The PUGS logic, however, will not allow a reset at point sensor #9 unless both #7 and #8 are uncovered. Therefore, since the reset at #9 was so large it is concluded that probably #7 and #8 were failed in the "uncovered" position.

The time difference between the uncovery of the oxidizer and fuel #9 point sensors (the oxidizer was approximately 7 seconds later than the fuel) confirms that there was an unbalance in the sump tanks, with the oxidizer quantity being approximately 1.0% greater than the fuel.

When the fuel point sensor #10 uncovered the reset caused the indicated unbalance to exceed the critical propellant unbalance limit and the C&W light was again activated at approximately 174 seconds after ignition. When the PUGS was switched back to the normal mode, the unbalance again exceeded the critical limit because of the erroneous oxidizer storage tank reading and the eighth C&W light activation occurred.

Because of the unexpected PUGS operation during the third burn the PUGS was inactivated for the fourth, fifth, and sixth SPS maneuvers. A PUGS self-test (engine off) test at approximately 125 hrs GET indicated satisfactory operation of all servo loops and the caution and warning system. The PUGS was activated for the seventh SPS maneuver, with the PUGS mode selection switch set in the "primary" position. Prior to the burn the oxidizer indicated quantity was adjusted to 10.8% using the test 2 switch. The resulting indicated fuel quantity was 15.4% because of the differences in the preset oxidizer and fuel slew rates when using the test 2 switch. As expected, a C&W light activation occurred following ignition of the seventh burn. The T/M data showed that the erroneous oxidizer storage tank probe reading was still present, and was 2.5 to 3.0% throughout the seventh burn. At the end of the burn the indicated sump tank unbalance showed approximately 2.2% (530 lbm) more oxidizer on board than fuel, indicating that the average mixture ratio for the first seven burns was about 2.5% less than the nominal 1.6.

In summary, it is concluded that the PUGS operation throughout the mission was satisfactory and as expected, with the exception of the erroneous oxidizer storage tank readings and the excessive level stabilization times. The PUGS will be disconnected from the Caution and Warning panel on future flights. The unbalance meter will still indicate the measured unbalance; however, the crews will be instructed to allow sufficient time (20 to 30 seconds) following ignition for the readings to stabilize before making any decision on PU valve position changes.

The erroneous oxidizer storage tank readings caused by the clinging propellant and drying dielectic compensator will be precluded by purposely calibrating the zero point of the probe to -0.4%. Although this may result in a small error when the storage tank has oxidizer in it, it will prevent errors of the 1.9 to 3.0% magnitude observed on this flight.

Prior to crossover it appeared that the oxidizer sump tank level was above the top of the sump tank primary probe, resulting in approximately 100 lbm of oxidizer being ungageable until after crossover. This condition was attributed to the transfer of oxidizer from the storage tank to the sump tank because of helium absorption in the sump tank during the time from propellant loading to the first SPS burn. Analytical calculations verify that such transfer will occur, although the exact amount depends on several variables such as the initial loads, the ullage pressure, and the percent saturation assumed. Therefore, it is reasonable to expect this oxidizer transfer on subsequent flights and to expect an error in both the indicated oxidizer quantity and propellant unbalance prior to crossover. The PU valve position change criteria to be used on G Mission, and subs, are being reviewed to determine how to best account for these errors.

7. PRESSURIZATION SYSTEM EVALUATION

Operation of the helium pressurization system was satisfactory without any indication of leakage. The helium supply pressure and the propellant ullage pressures indicated nominal helium usage for the eight SPS maneuvers.

The propellant tanks were pressurized to measured values of 179 psia for the oxidizer and 177 psia for the fuel several days prior to launch. As discussed previously, the fuel tank measurement was believed to be biased about -2 psi. There was little change in the tank pressures prior to launch, and at liftoff the measured tank pressures were 180 psia for oxidizer and 178 psia for fuel. Prior to launch of the Apollo 8 Mission, the oxidizer tank pressure increased approximately 9 psi becuase of heat input from the fuel cell heaters located in the top of Sector 4. Higher service module gaseous purge flowrates were employed on Apollo 9 which reduced the oxidizer tank pressure increase prior to launch by reducing the heat input to the ullage.

During the launch phase and coast period to the first SPS burn, the measured oxidizer and fuel tank pressures both decayed to approximately 175 psia, and were therefore relatively close the expected value of 178 psia at ignition of the first SPS firing.

During the coast following the second SPS burn, the measured oxidizer tank pressure increased approximately 6 psia and was approximately 182-183 psia at ignition of the third SPS burn. A similar, although larger (11 psia), increase was observed following the LOI-1 burn during the Apollo 8 Mission. These increases occur following burns where there is a significant percentage increase in ullage volume (the first long duration burn of a fully loaded SPS), and are attributed to propellant vapor resaturation and temperature recovery of the ullage.

8. ENGINE TRANSIENT ANALYSIS

A summary of the start and shutdown transient performance data for the eight SPS firings is presented in Table 8. No impulse or transient time values were calculated for the sixth burn because the short duration of the burn (1.4 seconds) precluded a good determination of the steady-state chamber pressure upon which to base such calculations. The transient times for both start and shutdown on all the seven burns analyzed were within their respective specification limits. The start impulse values computed for the fifth and seventh burns were greater than the upper specification limit of 700 lbf-sec by 24 lbf-sec and 291 lbf-sec, respectively, and, as seen in Table 8, the variability of the start impulse values exceeded the specified run-to-run tolerances of ± 200 lbf-sec. However, the good agreement between the times from ignition to 90% thrust on each burn indicates consistent performance. The computed shutdown impulse values were within the specified limits for each of the seven burns presented. The shutdown impulse variability between the burns, however, was greater than the specified run-to-run tolerances of ± 500 lbf-sec. The times from FS-2 to 10% thrust, however, showed consistent shutdown transient performance. The fact that the computed start and shutdown impulse values were not all within the specified engine-to-engine or run-to-run tolerances is not considered significant because of the inaccuracies associated with computing these values from the noisy flight chamber pressure data, and because the transient time values were all satisfactory.

The engine was started in the single bore mode (engine valve bank A) on all maneuvers. The chamber pressure overshoot values are contained in Table 8 and were less than the specified maximum of 120% on all eight starts. On all but the first and sixth burns, the remaining bore (engine valve bank B) was opened 3 to 4 seconds after ignition.

The first and sixth burns were conducted completely in the single bore mode. The GN_2 actuation system pressures indicated satisfactory usage. GN_2 System A was pressurized to 2480 psia at 68°F, and System B was pressurized to 2460 psia at 68°F during pre-launch servicing. Following the eighth burn the T/M data indicated that the System A pressure was 1950 psia and that the System B pressure was 2050 psia. There were eight burns which utilized System A for an indicated average usage of approximately 67 psi per burn. System B was utilized on six burns for an indicated average usage of approximately 68 psi per firing.

9. SPS THERMAL CONTROL

All Service Propulsion System temperatures remained within their redline limits throughout the mission. No SPS heater operation was required during the flight as engine and system line temperatures and the engine bipropellant valve temperature were maintained well within their limits using only passive thermal control. The engine injector flange temperature during all SPS firings was well below the 480°F redline limit. The maximum and minimum temperatures obtained from the flight data are contained in Table 9.

10. REFERENCES

1. NASA/MSC Memorandum, "Revision to Mission "D" SPS (Service Propulsion System) Preflight Analysis," from Joseph G. Thibodaux, Jr., dated 17 October 1968.

2. MSC-PA-R-69-2, "Apollo 9 Mission Report," dated May 1969.

3. TRW Technical Report No. 11176-14178-RO-00, "Apollo 8 CSM 103 Service Propulsion System Final Flight Evaluation," Prepared by R. J. Smith, dated 4 April 1969.

4. North American Aviation SID 66-1501-A, "Performance Data Supplement To Mission Modular Data Book, Block II Earth Orbital Mission, Revised 15 March 1967.

TABLE 1

SPS DUTY CYCLE

Burn	FS-1[1]	FS-2[1]	Burn Duration (secs)	Velocity Change[2] (ft/sec)
SPS 1	21541.08	21546.30	5.22	36.6
SPS 2	79924.07	80034.36	110.29	850.5
SPS 3	91059.27	91339.0	279.73	2567.9
SPS 4	102281.38	102309.24	27.86	300.5
SPS 5	195972.27	196015.53	43.26	572.5
SPS 6	444306.97	444308.40	1.43	33.7
SPS 7	610740.37	610765.26	24.89	650.1
SPS 8	865874.84	865886.58	11.74	322.7
		TOTAL	504.42	

(1) All times are from command module computer downlink data (CG0001V) except FS-2 time for SPS 3 which is from engine shutoff valve position (SP0022H) since SPS-3 cutoff was controlled by the entry monitor system. Times are G.E.T. seconds.

(2) Reference 2

TABLE 2

CSM 104 SPS ENGINE AND FEED SYSTEM CHARACTERISTICS

Engine No.	62
Injector No.	120
Chamber No.	346
Initial Chamber Throat Area (in.^{2})	121.6021
Oxidizer Engine Feedline Resistance (lb_f-sec^2/lb_m-ft^5)	488.7
Fuel Engine Feedline Resistance (lb_f-sec^2/lb_m-ft^5)	872.5
Oxidizer System Feedline Resistance (lb_f-sec^2/lb_m-ft^5)	97.72
Fuel System Feedline Resistance (lb_f-sec^2/lb_m-ft^5)	36.02

Characteristic Equation for C*:

$$C^* = C^*_{S.C.} + 870.5\,(MR - 1.6) - 273.83\,(MR^2 - 2.56) - 0.31878\,(P_C - 99) + 12.953\,(TP - 70) - 0.07414\,(TP^2 - 4900) - 5.466\,(MR \cdot TP - 112) + 0.03119\,(MR \cdot TP^2 - 7840.)$$

; where $C^*_{S.C.}$ (Engine No. 62) = 5934.3 ft/sec

Characteristic Equation for I_{SP}:

$$I_{SP} = I_{SP_{vac}} - 96.954\,(1.6 - MR) - 0.0487\,(99 - P_C) - 0.06276\,(70 - TP) + 30.409\,(2.56 - MR^2) + 0.0004483\,(4900 - TP^2)$$

; where $I_{SP_{vac}}$ (Engine No. 62) = 313.0 lb_f-sec/lb_m

TABLE 3

SUMMARY OF MEASURED STEADY-STATE PRESSURE DATA

MEASUREMENT	BURN									
	1	2		3		4	5	6	7	8
	FS1+4 sec.	FS1+20 sec.	FS1+100 sec.	FS1+20 sec.	FS1+220 sec.	FS1+12 sec.	FS1+23 sec.	FS1+1.5 sec.	FS1+13 sec.	FS1+9 sec.
SP0003P Oxidizer Tank	174	175	175	174	176	175	175	176	175	174
SP0931P Oxidizer Interface	160	159	159	158	161	163	161	161	162	160
SP0006P Fuel Tank	174	173	173	174	175	177	174	172	174	175
SP0930P Fuel Interface	171	170	170	169	172	175	172	171	172	172
SP0661P Engine Chamber	96	100	102	102	104	106	102	99	103	102

NOTE: All pressures are in psia.

TABLE 4

FLIGHT DATA USED IN STEADY STATE ANALYSIS

Measurement Number	Description	Range	Sample Rate Samples/Sec
SP0930 P	Pressure, Engine Fuel Interface	0 to 300 psia	10
SP0931 P	Pressure, Engine Oxidizer Interface	0 to 300 psia	10
SP0661 P	Pressure, Engine Chamber	0 to 150 psia	100
SP0003 P	Pressure, Oxidizer Tanks	0 to 250 psia	10
SP0006 P	Pressure, Fuel Tanks	0 to 250 psia	10
SP0048 T	Temperature, Engine Fuel Feed Line	0 to 200 °F	1
SP0049 T	Temperature, Engine Oxidizer Feed Line	0 to 200 °F	1
SP0054 T	Temperature, 1 Oxidizer Distribution Line	0 to 200 °F	1
SP0057 T	Temperature, 1 Fuel Distribution Line	0 to 200 °F	1
SP0655 Q	Quantity, Oxidizer Tank 1 Primary - Total Auxiliary	0 to 50%	1
SP0656 Q	Quantity, Oxidizer Tank 2	0 to 60%	1
SP0657 Q	Quantity, Fuel Tank 1 Primary - Total Auxiliary	0 to 50%	1
SP0658 Q	Quantity, Fuel Tank 2	0 to 60%	1
CG0001 V	Computer Digital Data	40 Bits	1/2

TABLE 5

SERVICE PROPULSION SYSTEM STEADY-STATE PERFORMANCE

SECOND SPS BURN

PARAMETER	FS-1 + 90 Seconds		
	Predicted	Measured	Calculated
	INSTRUMENTED		
Oxidizer Tank Pressure, psia	176	174	175
Fuel Tank Pressure, psia	174	173	175
Oxidizer Interface Pressure, psia	162	158	161
Fuel Interface Pressure, psia	170	170	171
Engine Chamber Pressure, psia	100	101	100
	DERIVED		
Oxidizer Flowrate, lbm/sec	40.4	--	40.2
Fuel Flowrate, lbm/sec	25.5	--	25.9
Propellant Mixture Ratio	1.59	--	1.55
Vacuum Specific Impulse, sec	313.0	--	313.9
Vacuum Thrust, lbf	20626	--	20764

(1) Predicted values from Reference 1

(2) Calculated values from Propulsion Analysis Program

(3) Measured data are as recorded and are not corrected for biases and errors discussed in text.

TABLE 6

SERVICE PROPULSION SYSTEM STEADY-STATE PERFORMANCE

THIRD SPS BURN

PARAMETER	FS-1 + 225 Seconds		
	Predicted	Measured	Calculated
	INSTRUMENTED		
Oxidizer Tank Pressure, psia	177	176	175
Fuel Tank Pressure, psia	175	175	176
Oxidizer Interface Pressure, psia	165	162	164
Fuel Interface Pressure, psia	172	173	174
Engine Chamber Pressure, psia	102	104	102
	DERIVED		
Oxidizer Flowrate, lbm/sec	41.1	--	40.7
Fuel Flowrate, lbm/sec	25.6	--	26.2
Propellant Mixture Ratio	1.60	--	1.55
Vacuum Specific Impulse, sec	313.1	--	313.9
Vacuum Thrust, lbf	20885	--	21011

NOTES:

(1) Predicted values from Reference 1

(2) Calculated values from Propulsion Analysis Program

(3) Measured data are as recorded and are not corrected for biases and errors discussed in text.

TABLE 7

SPS PROPELLANT LOADING DATA

PROPELLANT	Total Mass Loaded (lbm)		
	Loading Data		Analysis[c] Results
	110 psia[b]	175 psia	
Oxidizer[a]	22247	22226	22143
Fuel[a]	13939	13882	13844
Total[a]	36186	36108	35987

[a]Includes gageable, ungageable, and vapor loaded quantities.

[b]Load reported by KSC in Spacecraft Operational Data Book.

[c]Loads based on extrapolation of second burn analysis results.

TABLE 8

ENGINE TRANSIENT DATA

PARAMETER	SPECIFICATION VALUE		APOLLO 9 SPS MANEUVERS							
	Single Bore	Dual Bore	1st	2nd	3rd	4th	5th	6th	7th	8th
Total Vacuum Impulse (Ignition to 90% Steady-State Thrust), lbf-sec	450 +250 (a) ±200 (b)		221	398	685	678	724	N/A	991	513
Time (Ignition to 90% Steady-State Thrust), sec	0.675 ±0.100		.606	.617	.605	.576	.580	N/A	.640	.613
Chamber Pressure Overshoot, Percent	120		117	101	108	113	114	113 (psia)	115	113
Total Vacuum Impulse (cutoff to 0% steady-state thrust), lbf-sec	12,500 ± 2,500 (a) ± 500 (b)	13,500 ± 2,500 (a) ± 500 (b)	10,173	12,549	11,985	11,871	12,585	N/A	13,415	12,824
Time (Cutoff to 10% Steady-State Thrust), Sec	1.075 ±0.175	1.075 ±0.175	.906	.996	.922	.915	1.02	N/A	1.05	1.023

(a) Engine to Engine Tolerance
(b) Run to Run Tolerance

Note: Maneuvers 1 and 6 had single bore starts and shutdowns.
Maneuvers 2,3,4,5,7, and 8 had single bore starts and dual bore shutdowns.

TABLE 9

SPS TEMPERATURES

Measurement Number	Measurement Description	Temperatures (°F)			
		Minimum		Maximum	
		Redline	Actual	Redline	Actual
SP0045T	Engine Bipropellant Valve	40	55	160	138[3]
SP0048T	Fuel Engine Line	25[1]/40[2]	65	110	88
SP0049T	Oxidizer Engine Line	25[1]/40[2]	66	110	86
SP0054T	Oxidizer System Line	25[1]/40[2]	66	110	75
SP0057T	Fuel System Line	25[1]/40[2]	67	110	75
SP0061T	Engine Injector Flange No. 1	N/A	-	480[2]	240[4]
SP0062T	Engine Injector Flange No. 2	N/A	-	480[2]	230[4]

(1) Critical burns

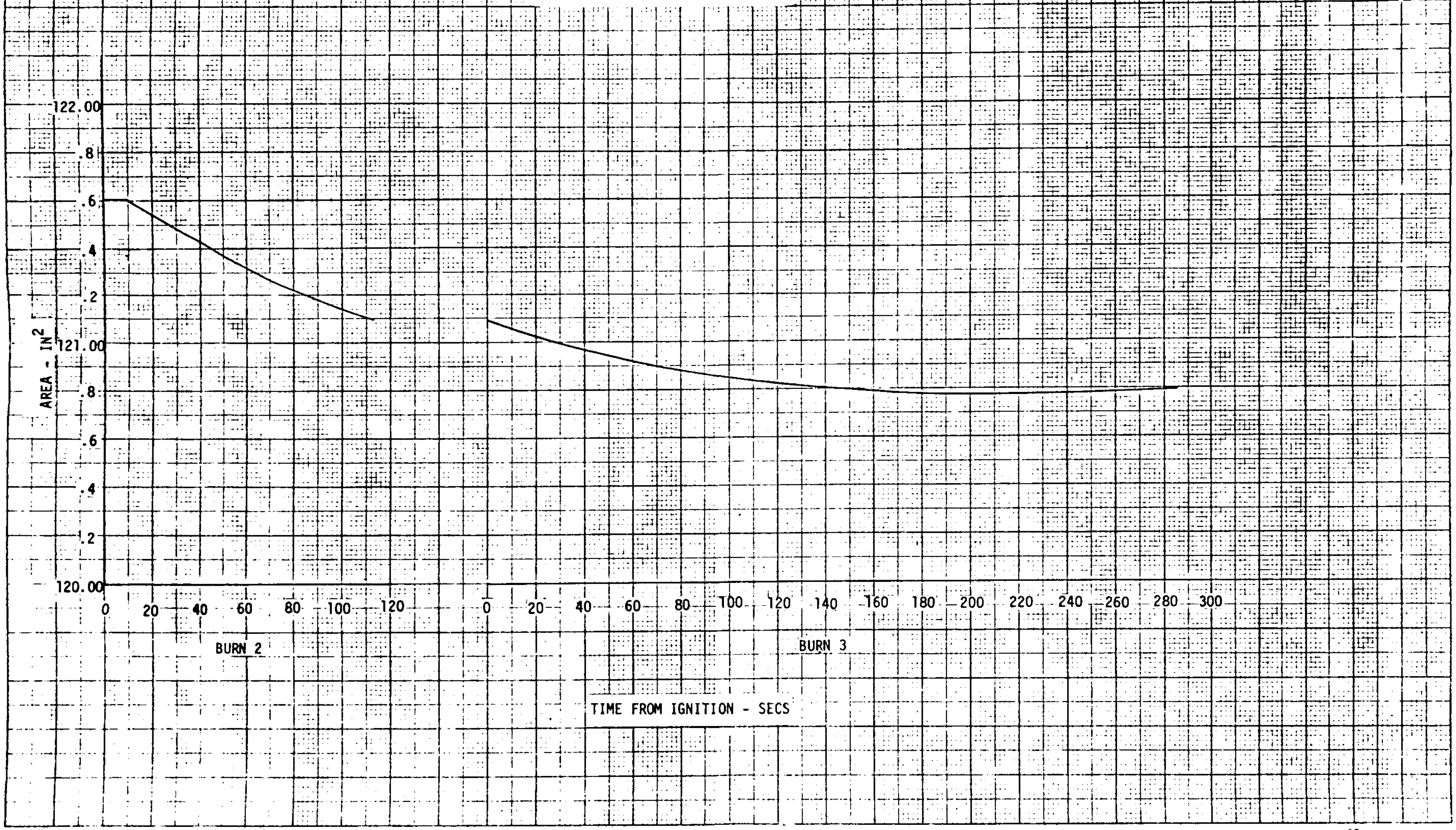
(2) Non-Critical Burns

(3) Maximum soakback following fourth SPS burn

(4) Maximum during fourth SPS burn

FIGURE 1

SPS CHAMBER THROAT AREA



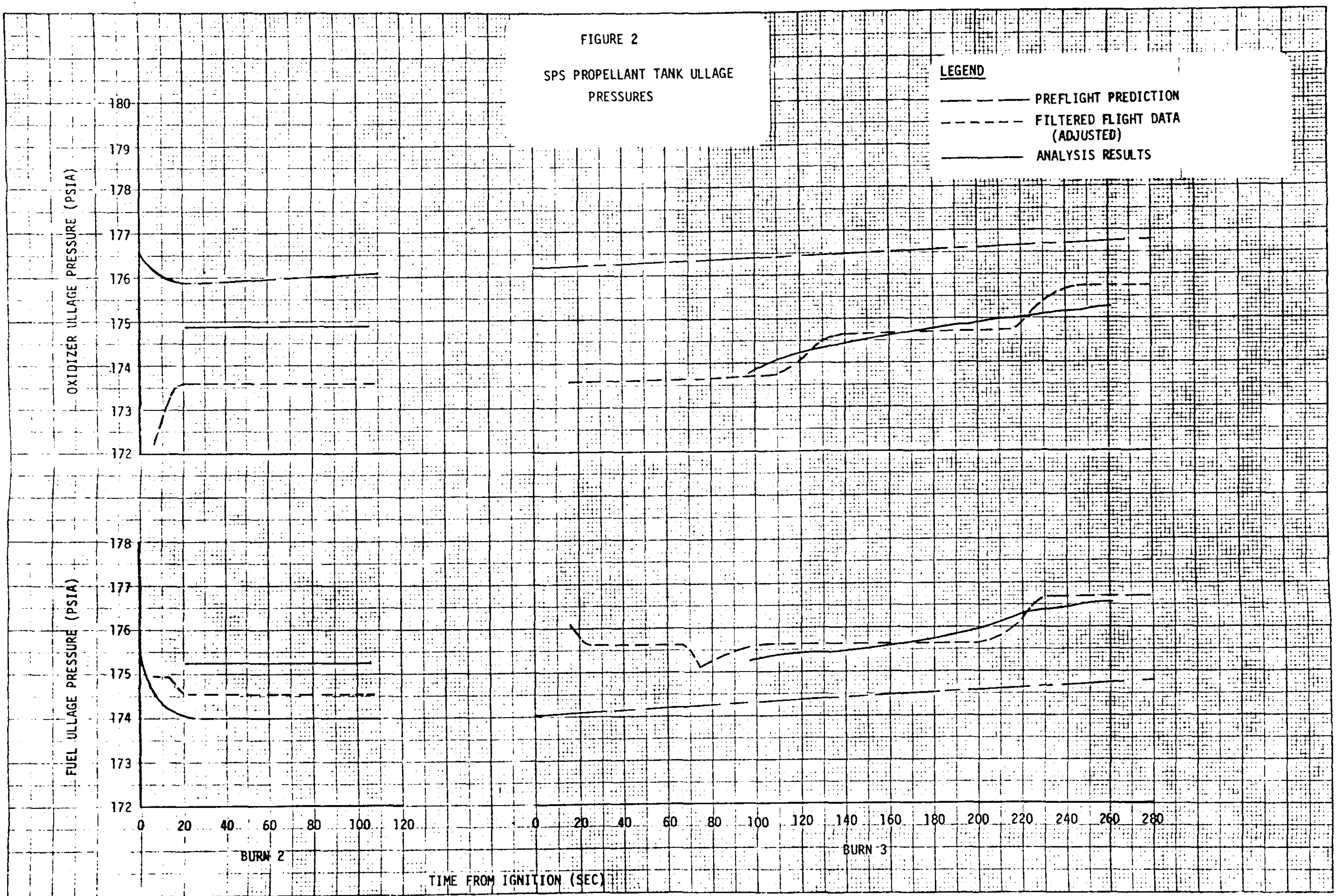
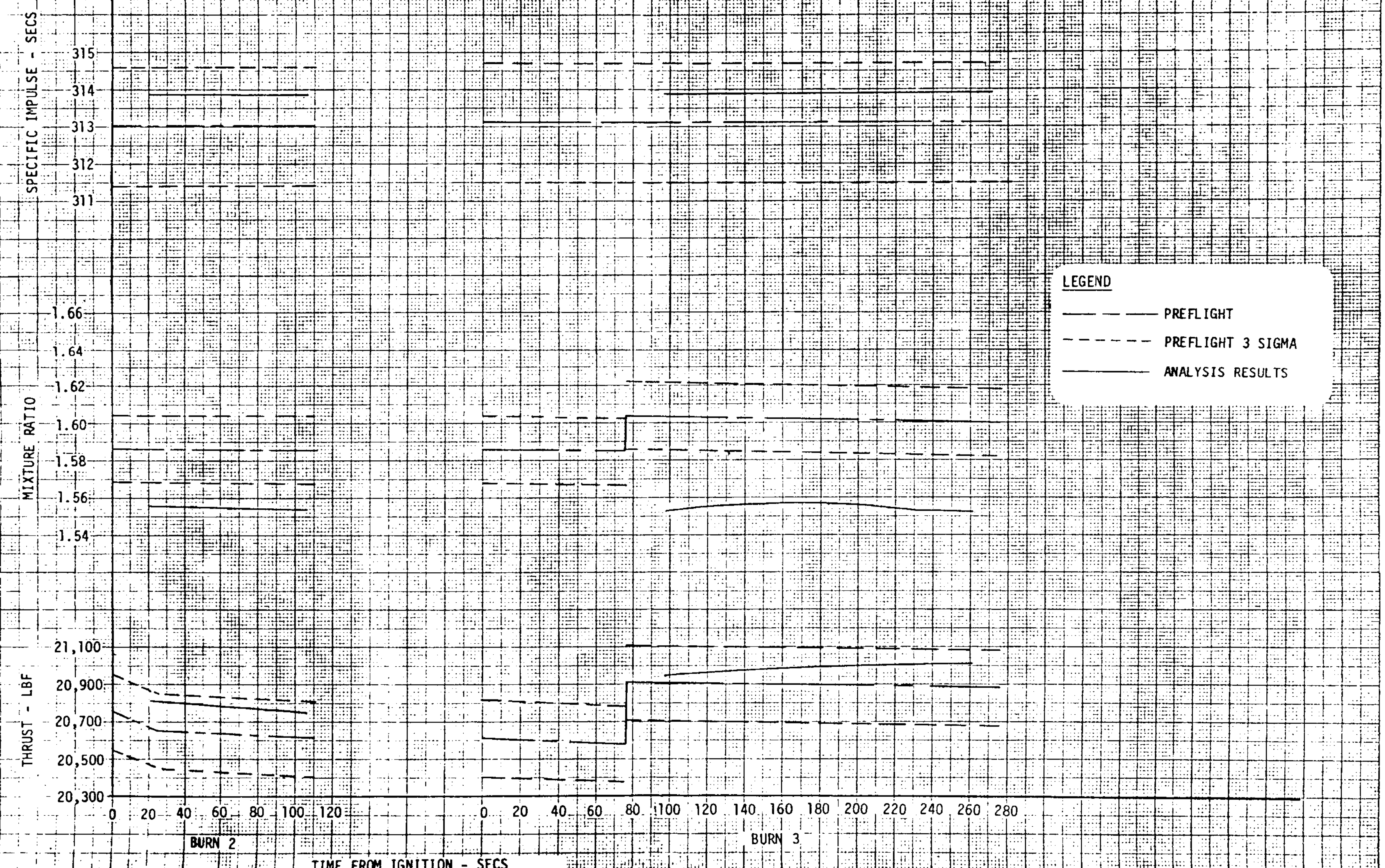


FIGURE 2
SPS PROPELLANT TANK ULLAGE PRESSURES
LEGEND
PREFLIGHT PREDICTION
FILTERED FLIGHT DATA (ADJUSTED)
ANALYSIS RESULTS
OXIDIZER ULLAGE PRESSURE (PSIA)
180
179
178
177
176
175
174
173
172
FUEL ULLAGE PRESSURE (PSIA)
178
177
176
175
174
173
172
0 20 40 60 80 100 120
BURN 2
0 20 40 60 80 100 120 140 160 180 200 220 240 260 280
BURN 3
TIME FROM IGNITION (SEC)

FIGURE 3

COMPARISON OF PREFLIGHT PREDICTED AND INFLIGHT PERFORMANCE



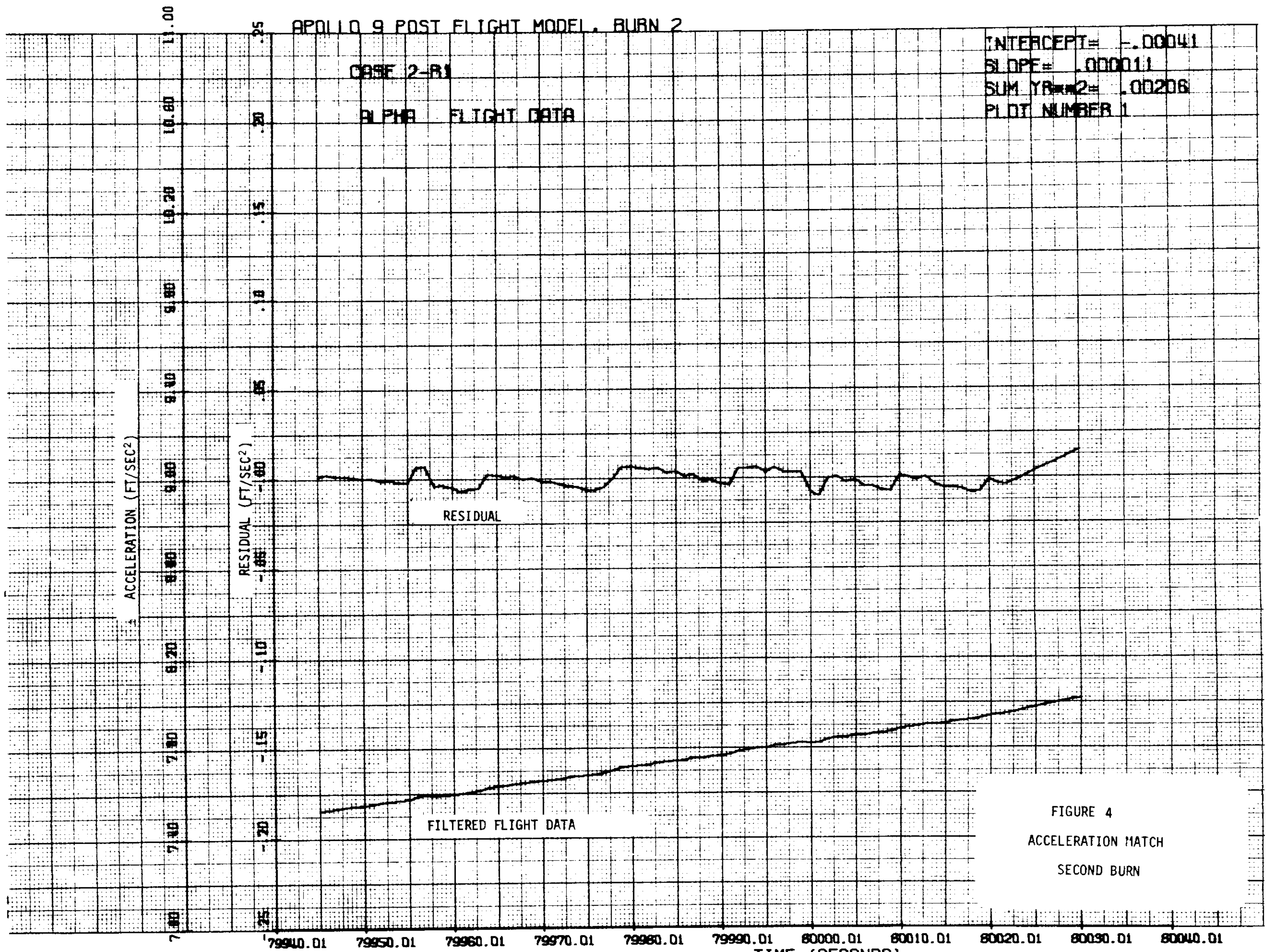


FIGURE 4
ACCELERATION MATCH
SECOND BURN

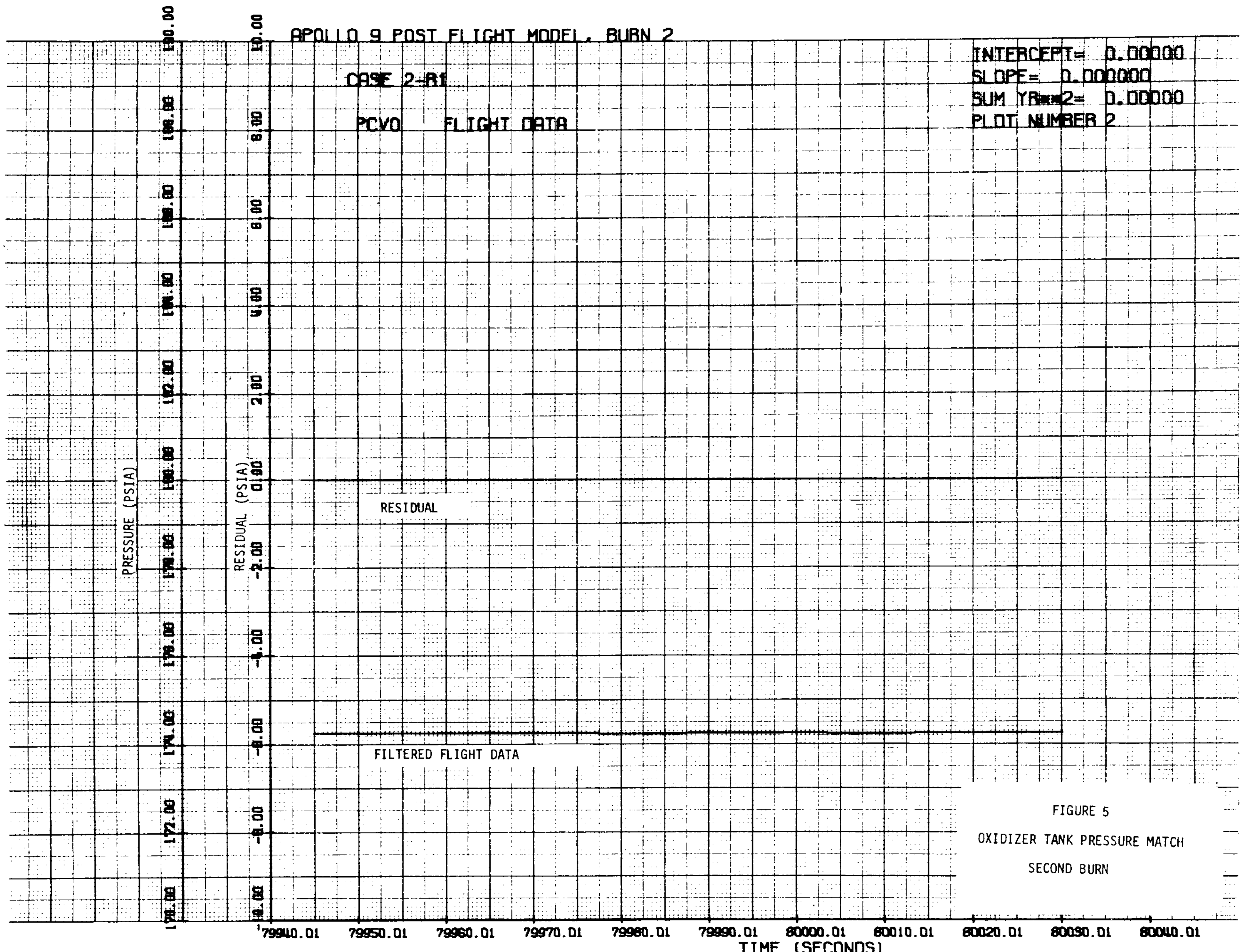


FIGURE 5
OXIDIZER TANK PRESSURE MATCH
SECOND BURN

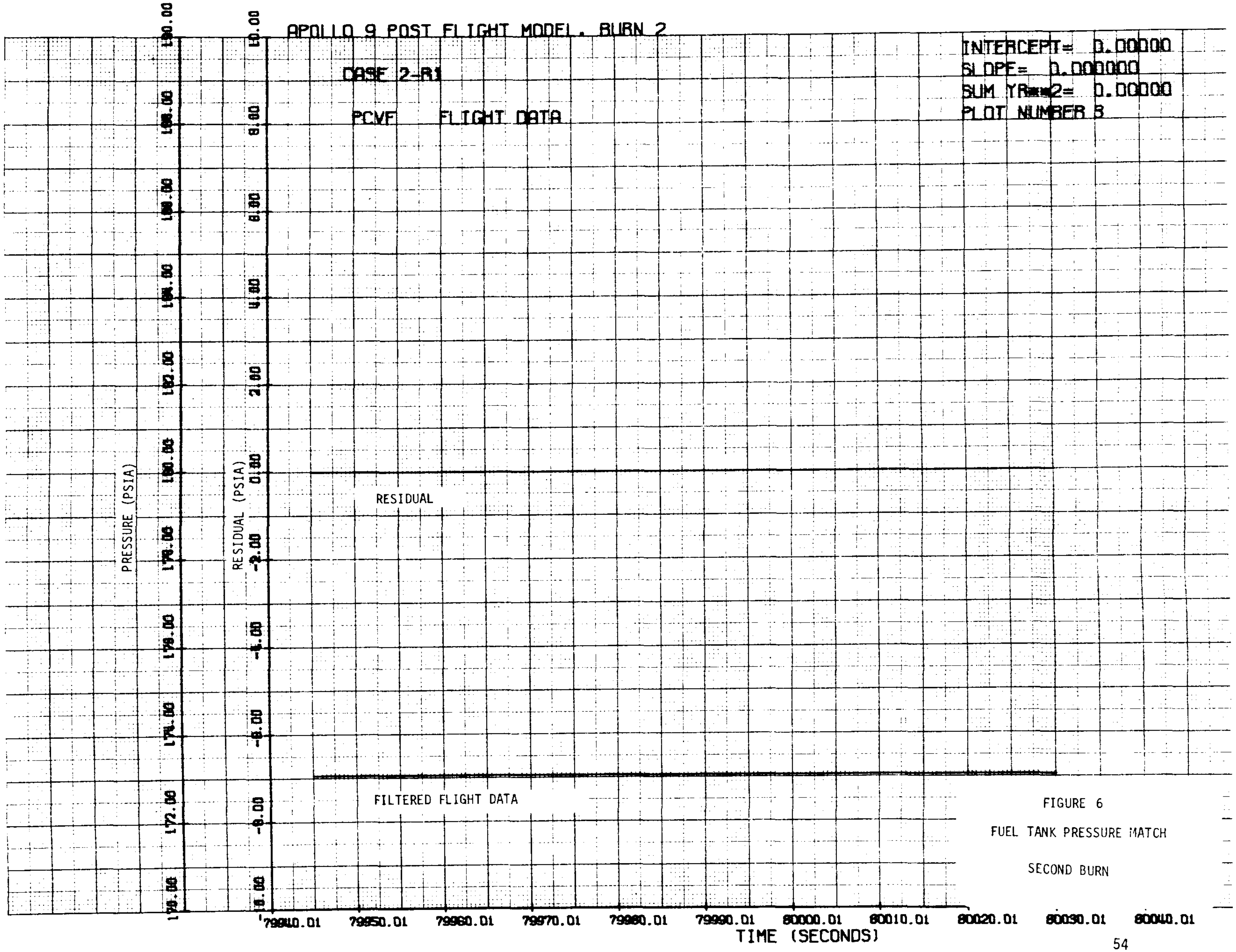


FIGURE 6
FUEL TANK PRESSURE MATCH
SECOND BURN

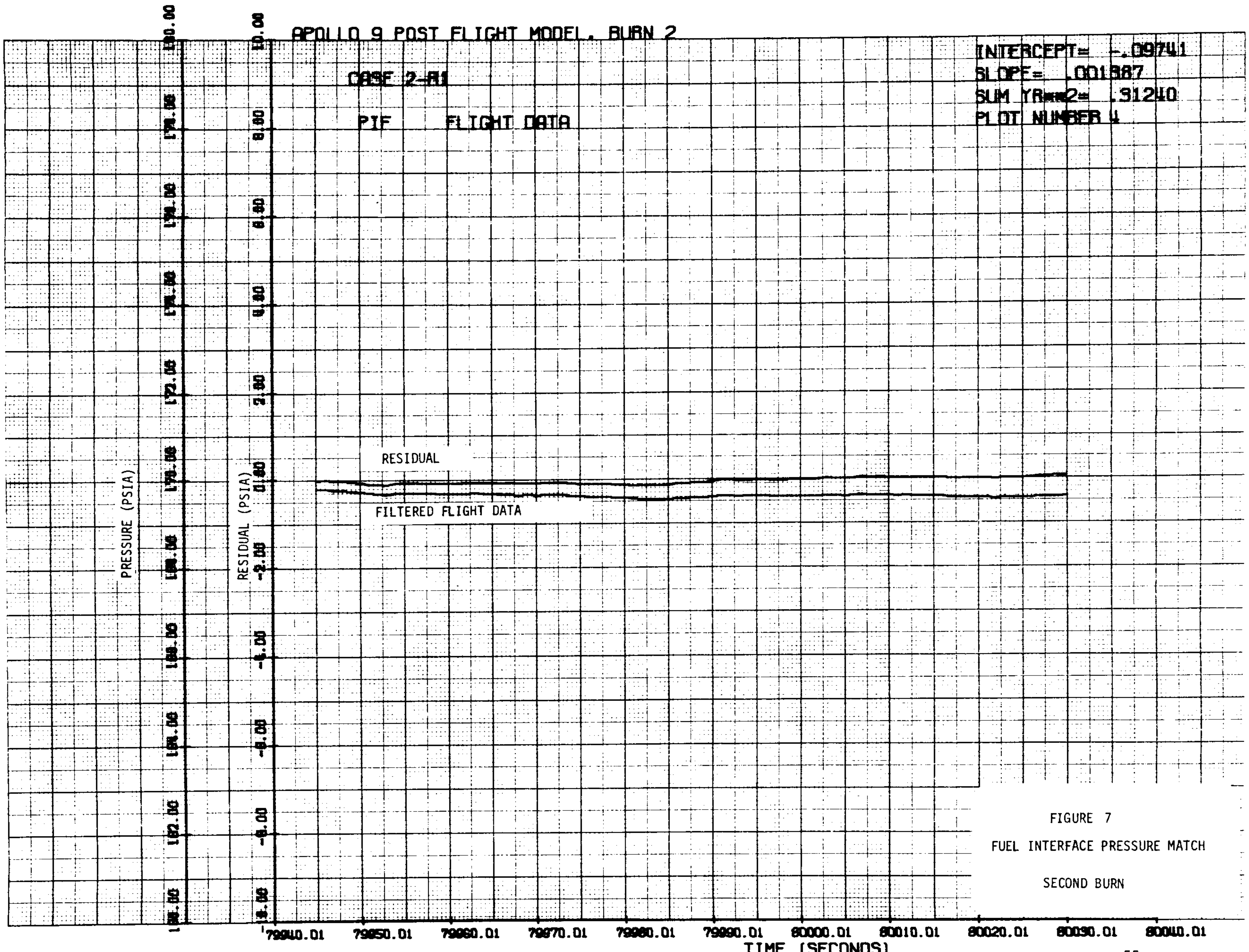


FIGURE 7

FUEL INTERFACE PRESSURE MATCH

SECOND BURN

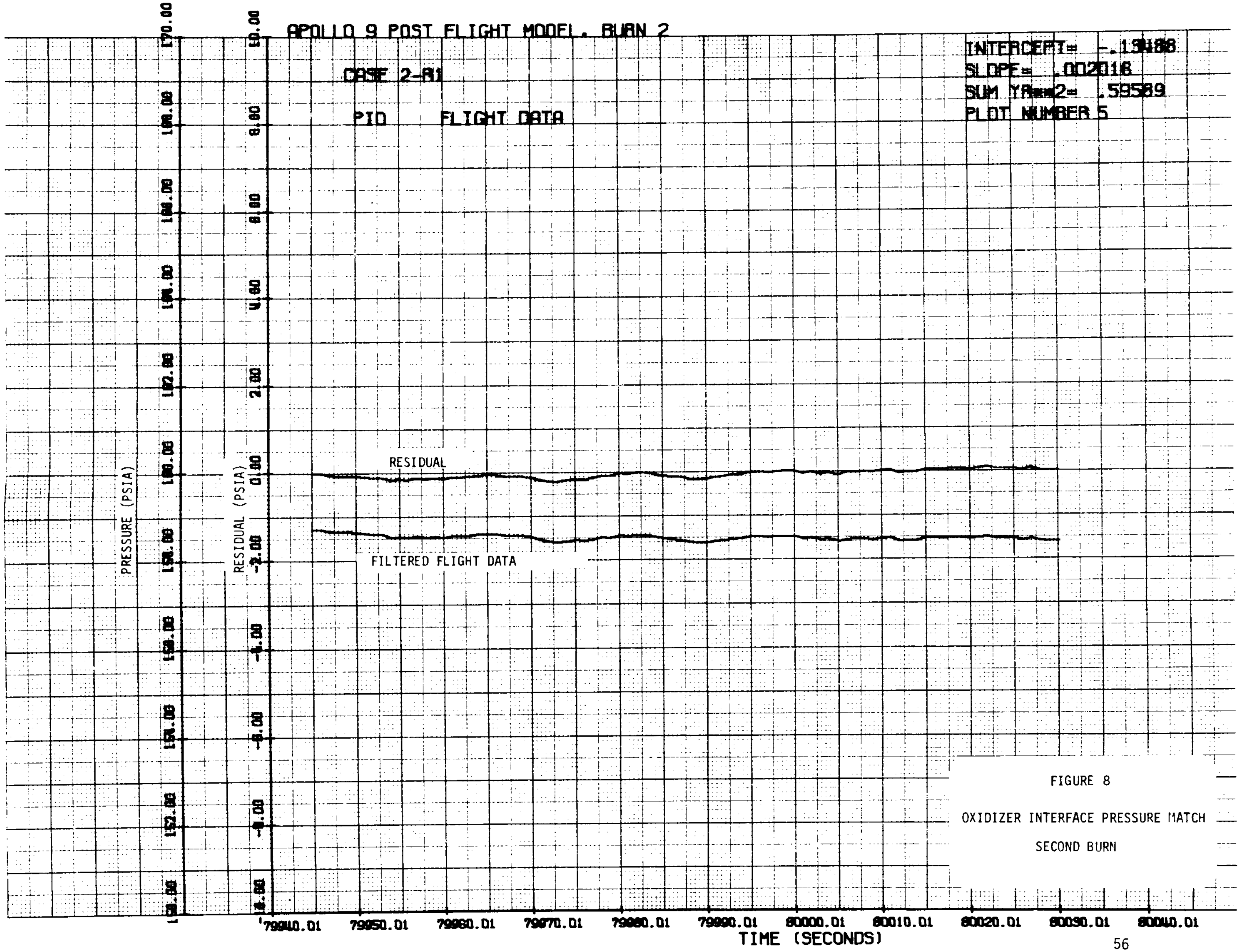


FIGURE 8

OXIDIZER INTERFACE PRESSURE MATCH

SECOND BURN

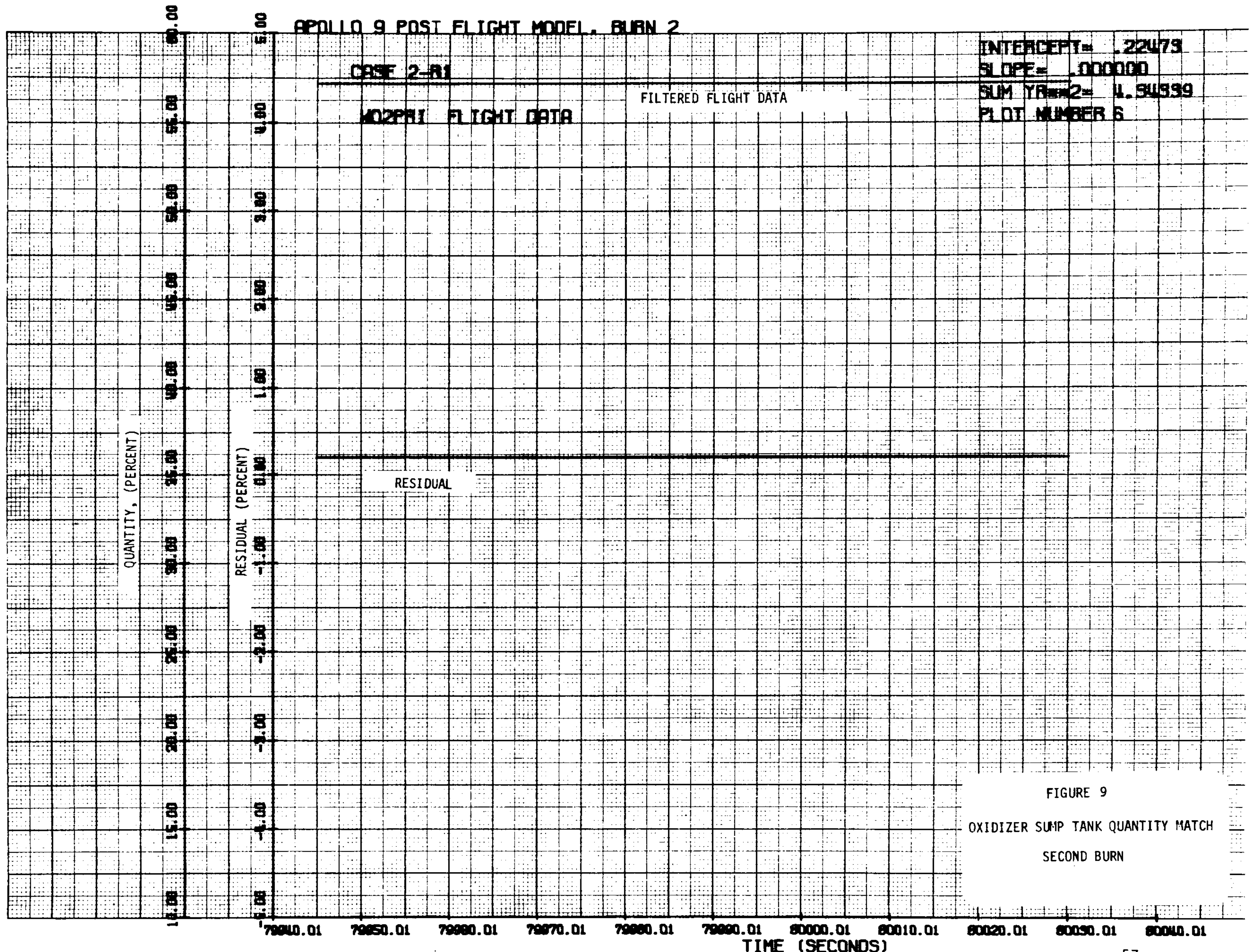


FIGURE 9

OXIDIZER SUMP TANK QUANTITY MATCH

SECOND BURN

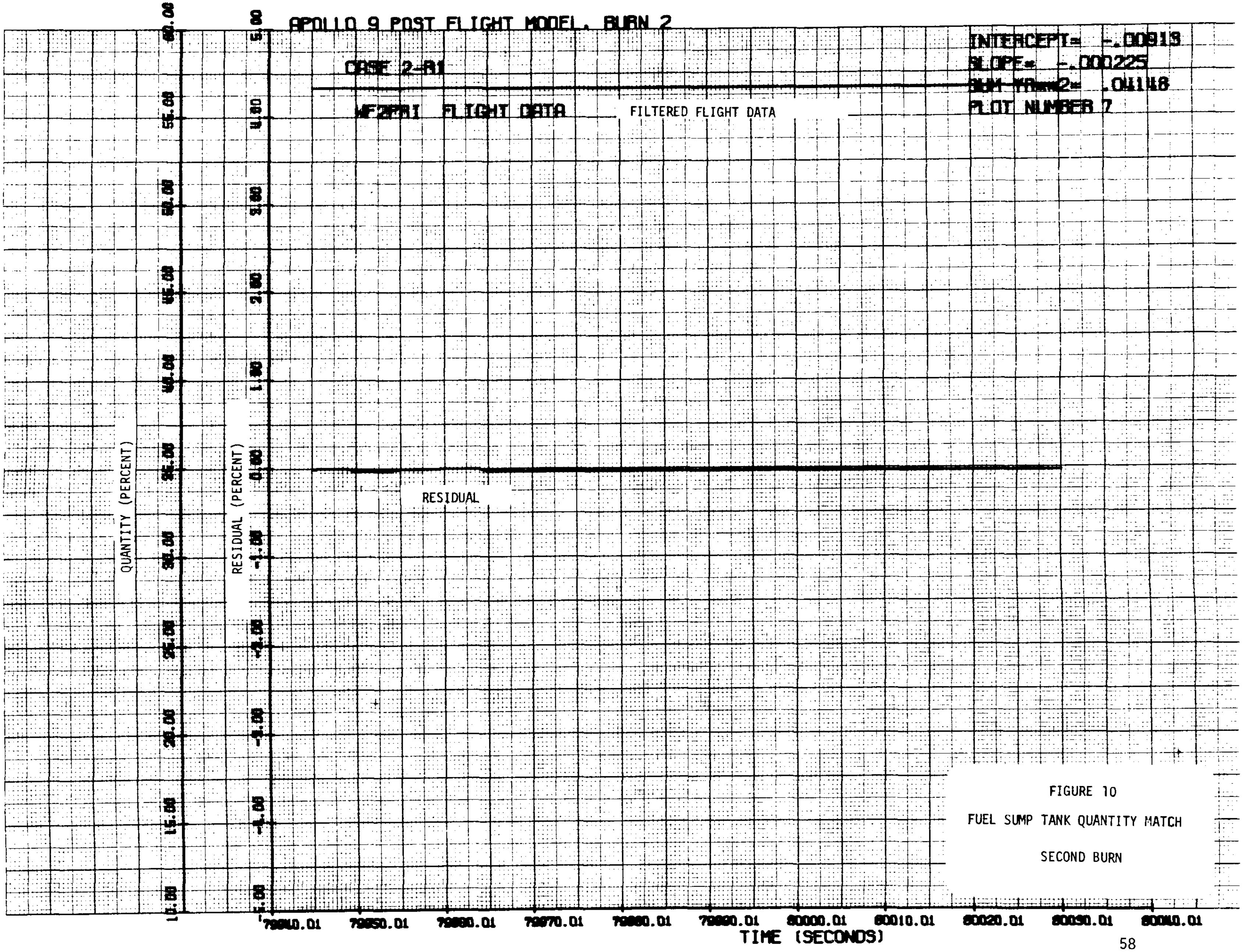


FIGURE 10

FUEL SUMP TANK QUANTITY MATCH

SECOND BURN

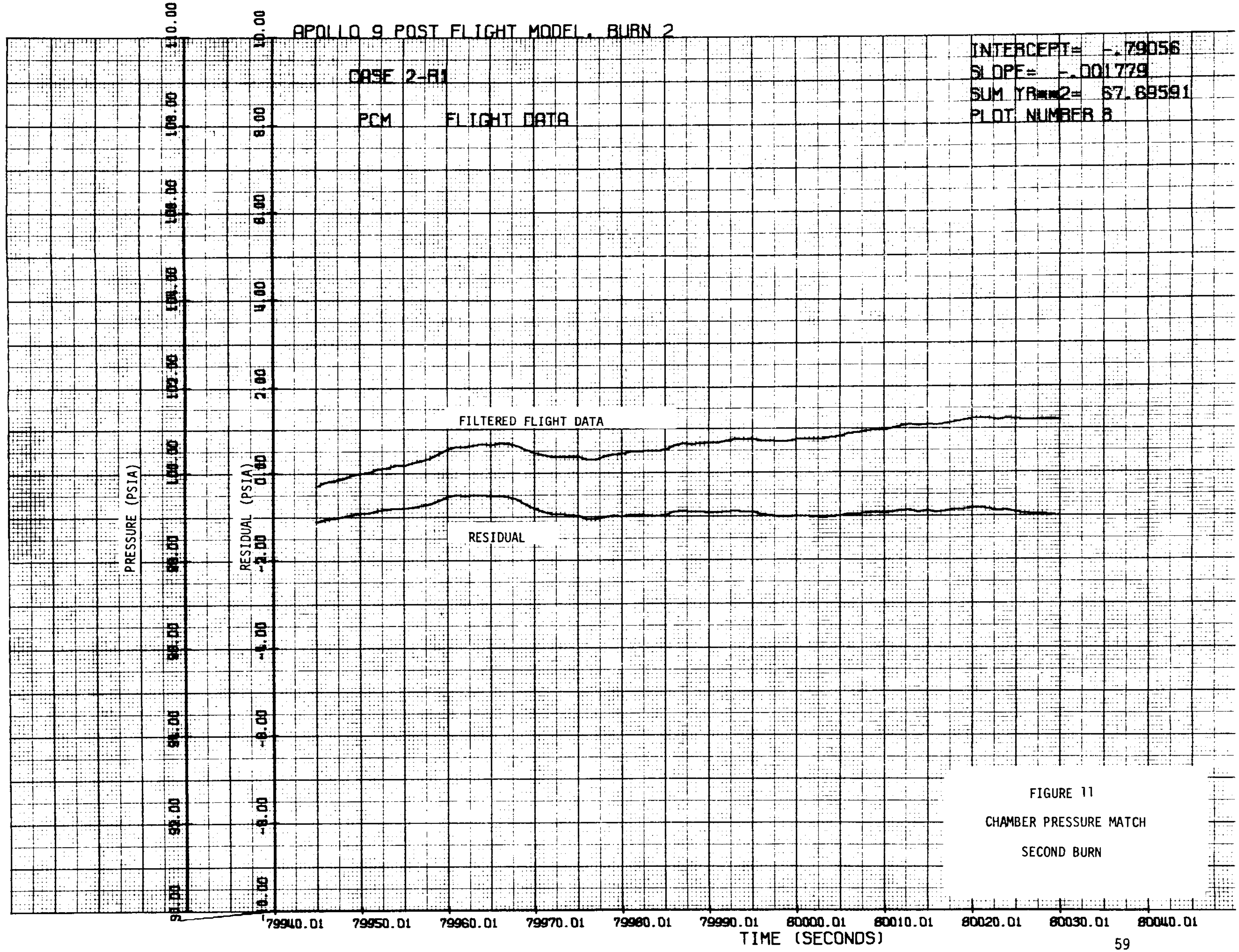


FIGURE 11

CHAMBER PRESSURE MATCH

SECOND BURN

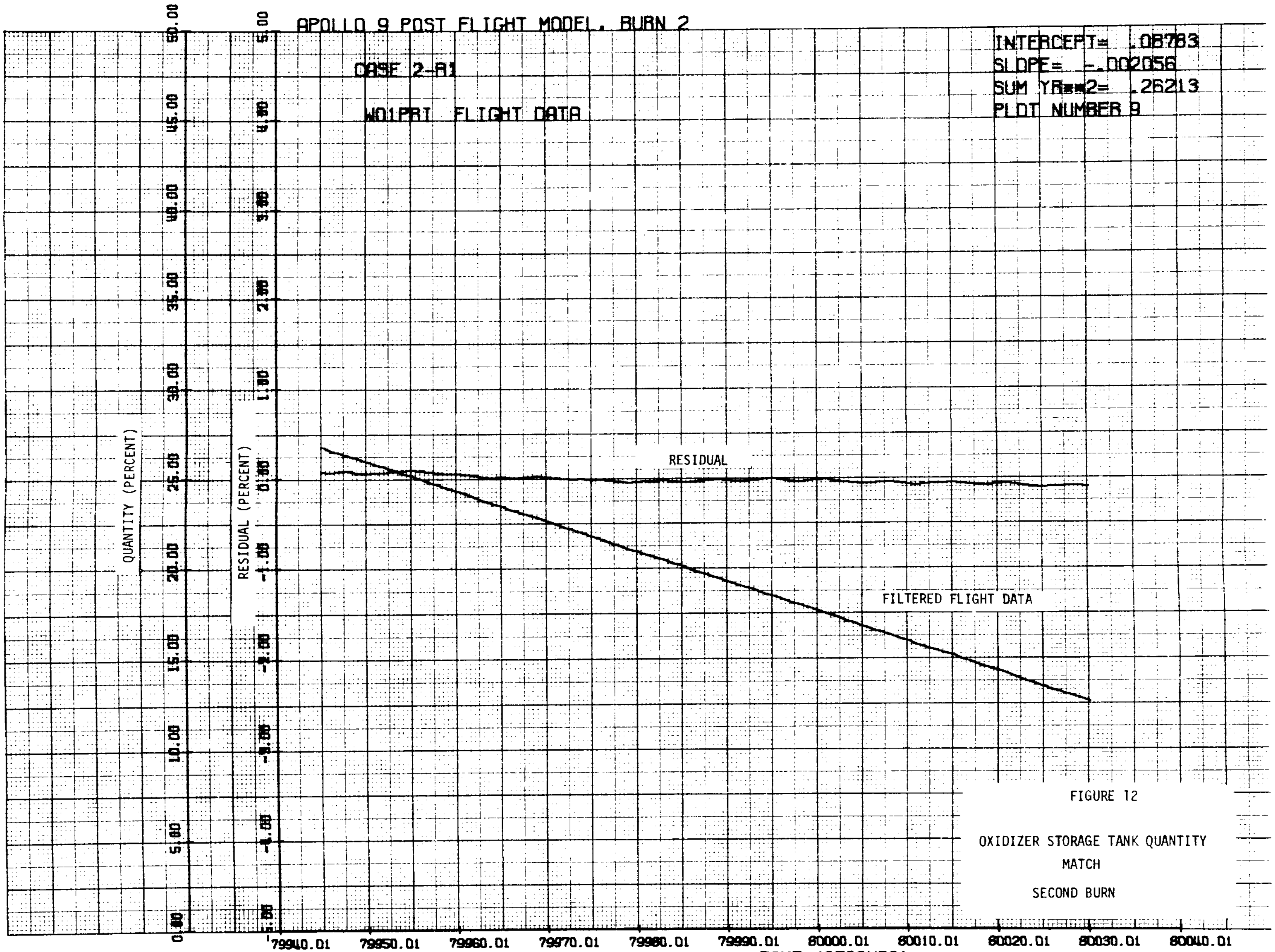


FIGURE 12

OXIDIZER STORAGE TANK QUANTITY
MATCH
SECOND BURN

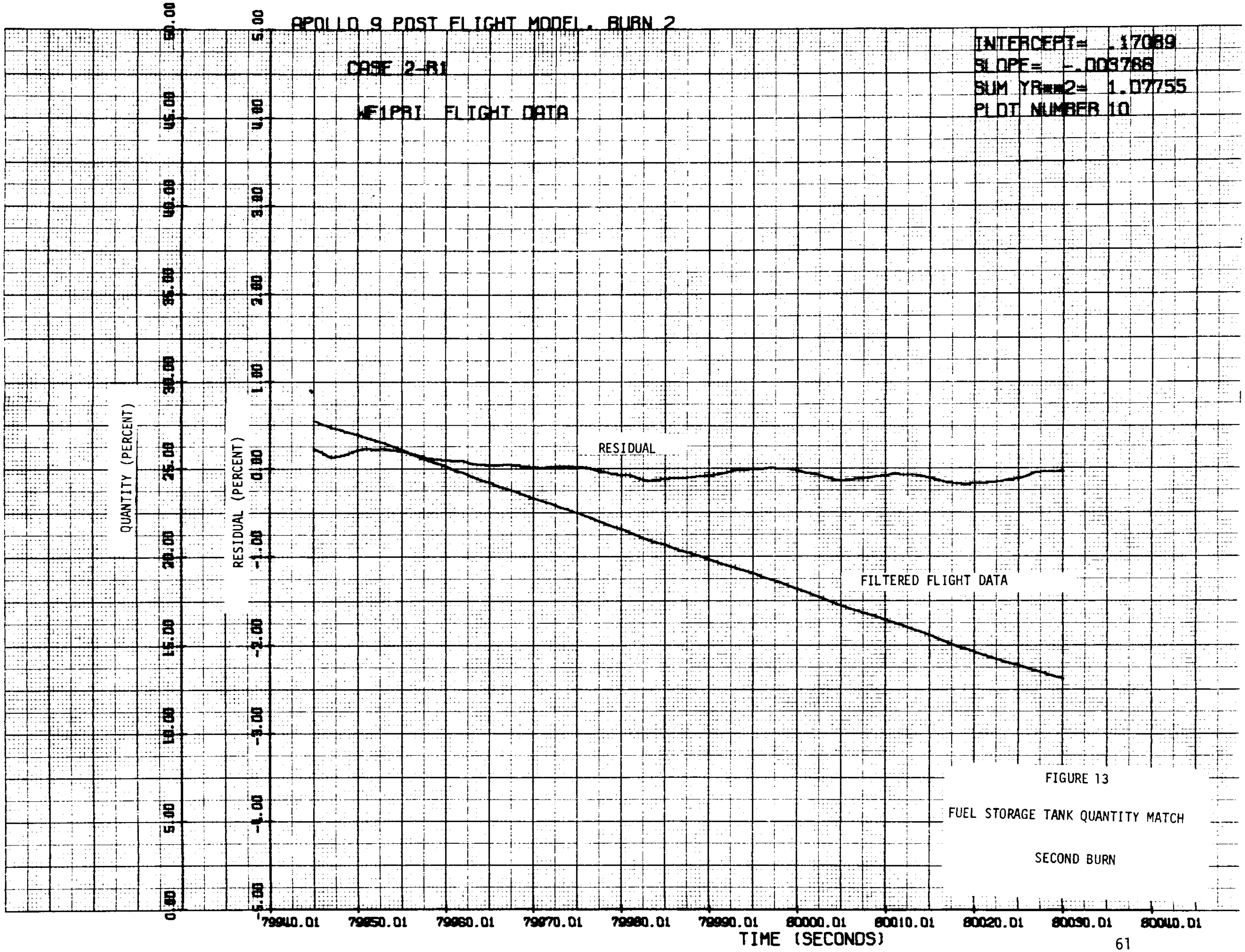


FIGURE 13

FUEL STORAGE TANK QUANTITY MATCH

SECOND BURN

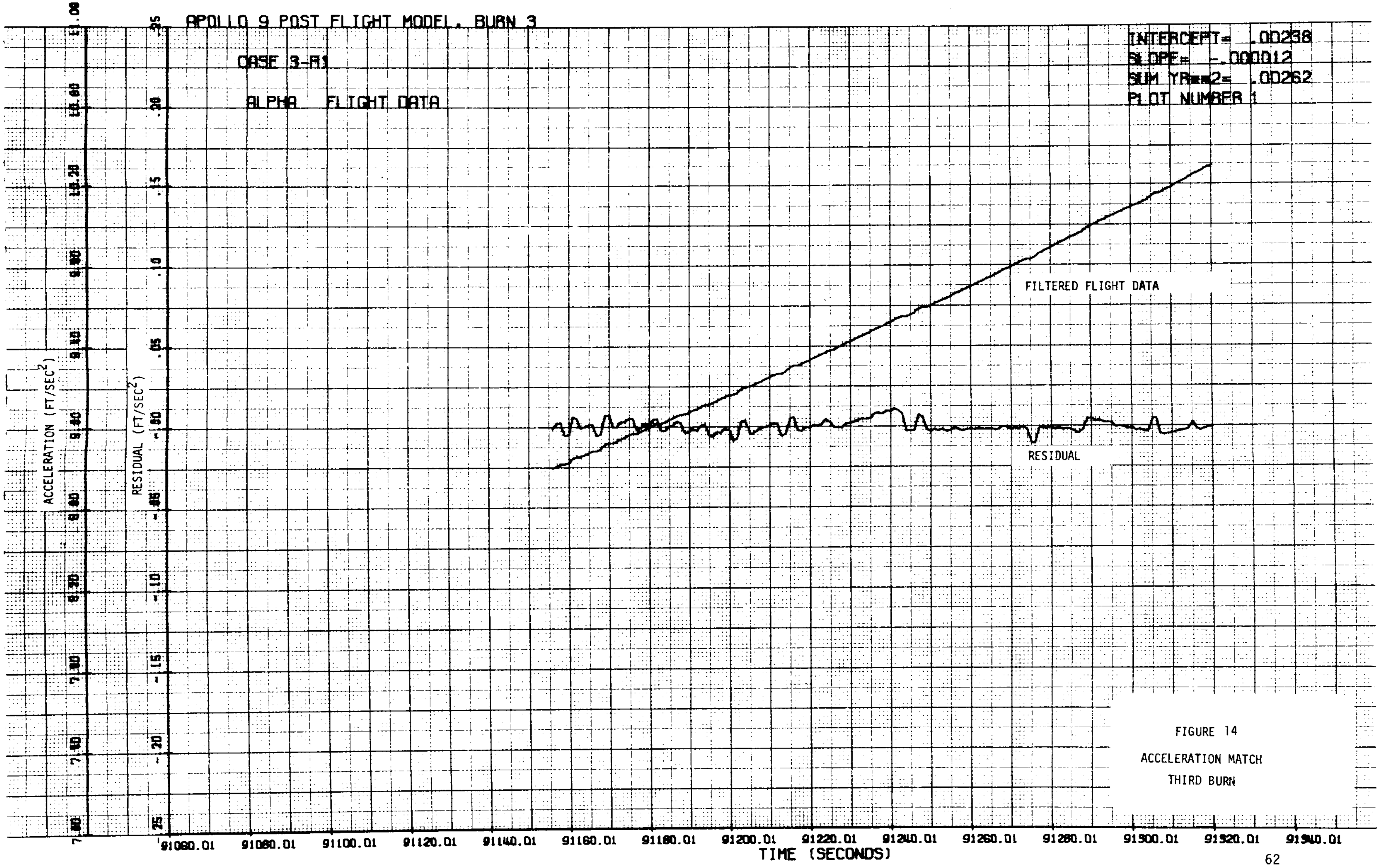


FIGURE 14
ACCELERATION MATCH
THIRD BURN

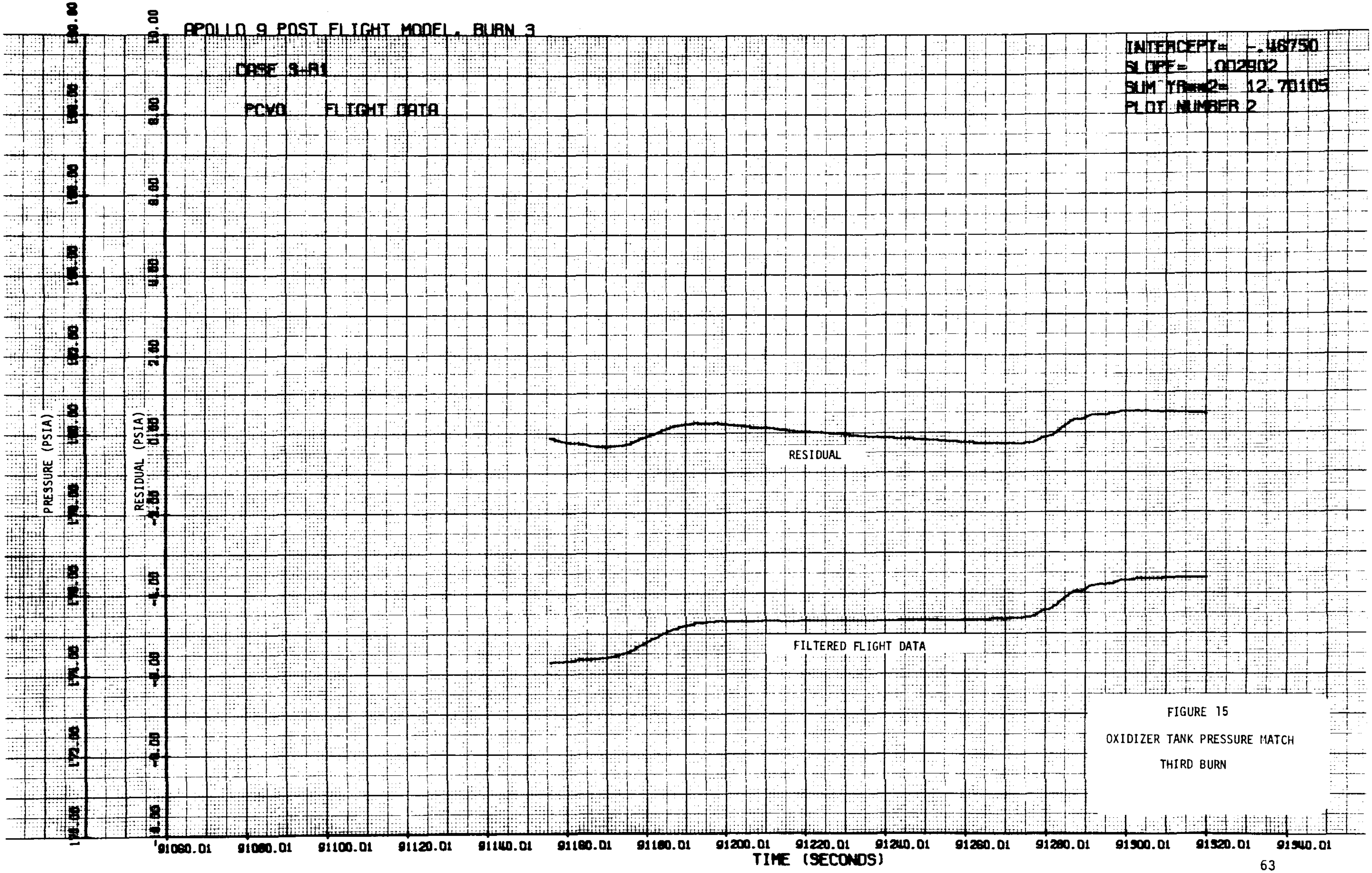


FIGURE 15

OXIDIZER TANK PRESSURE MATCH

THIRD BURN

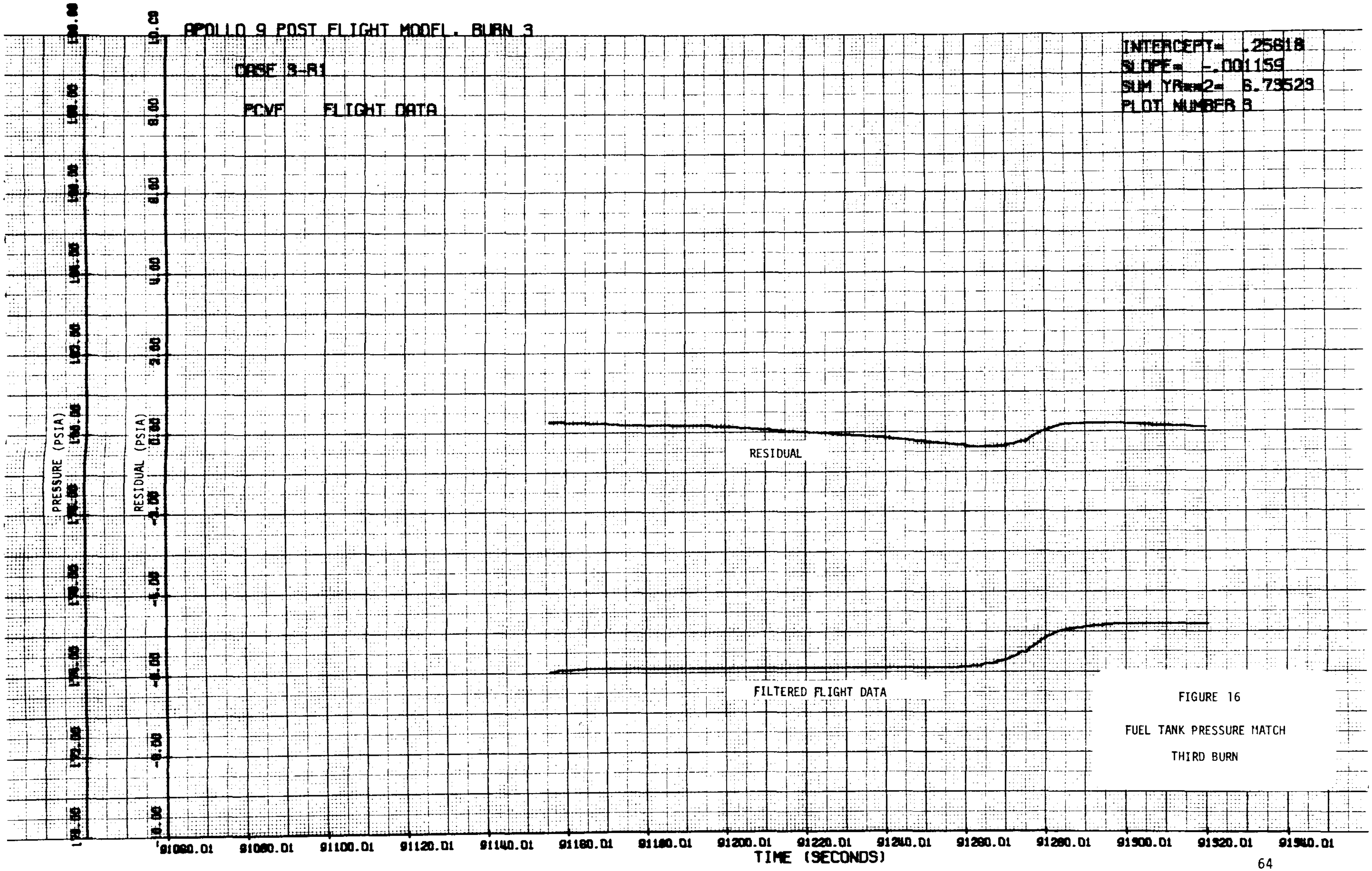


FIGURE 16

FUEL TANK PRESSURE MATCH

THIRD BURN

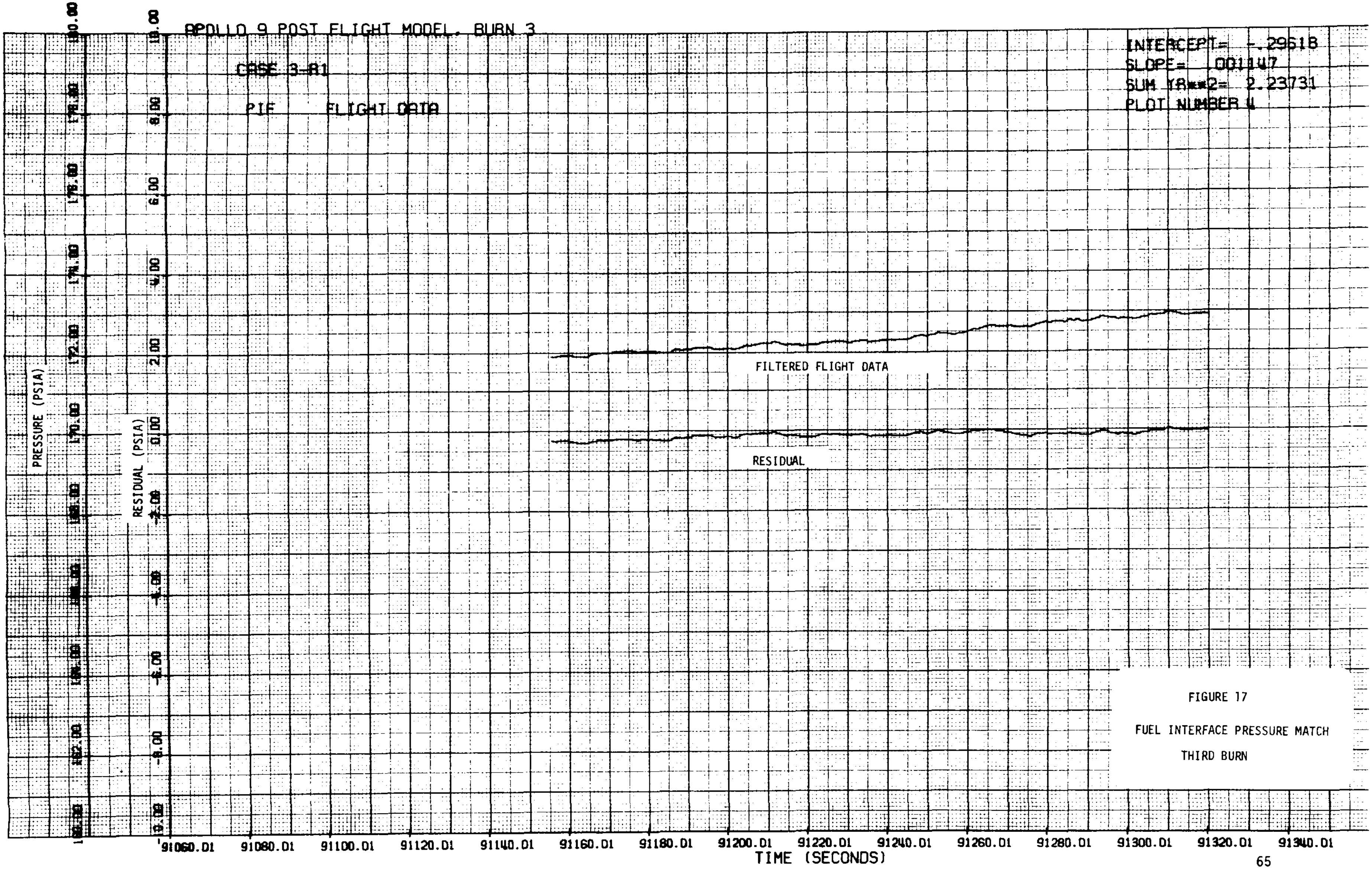


FIGURE 17

FUEL INTERFACE PRESSURE MATCH

THIRD BURN

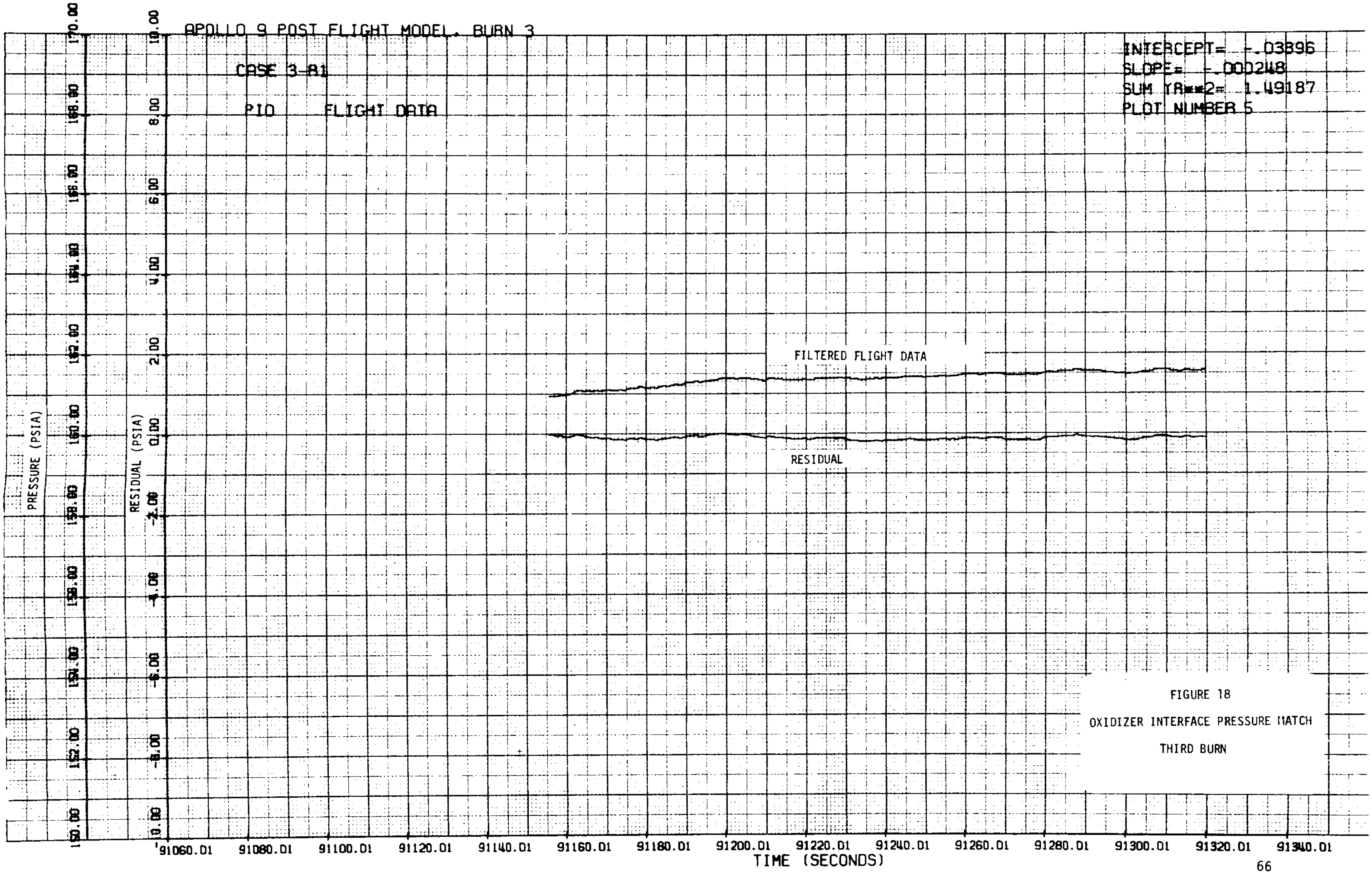


FIGURE 18

OXIDIZER INTERFACE PRESSURE MATCH

THIRD BURN

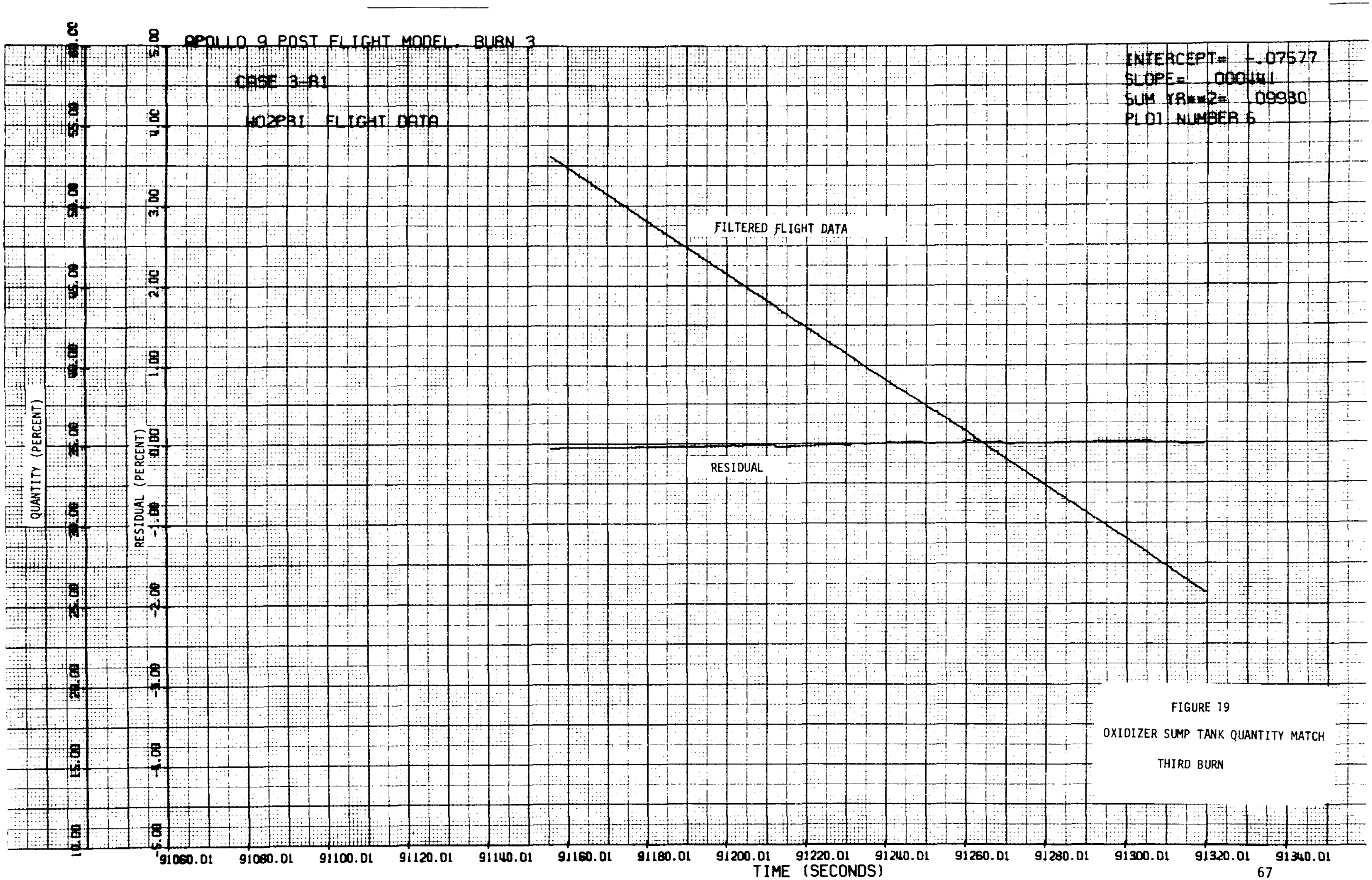


FIGURE 19

OXIDIZER SUMP TANK QUANTITY MATCH

THIRD BURN

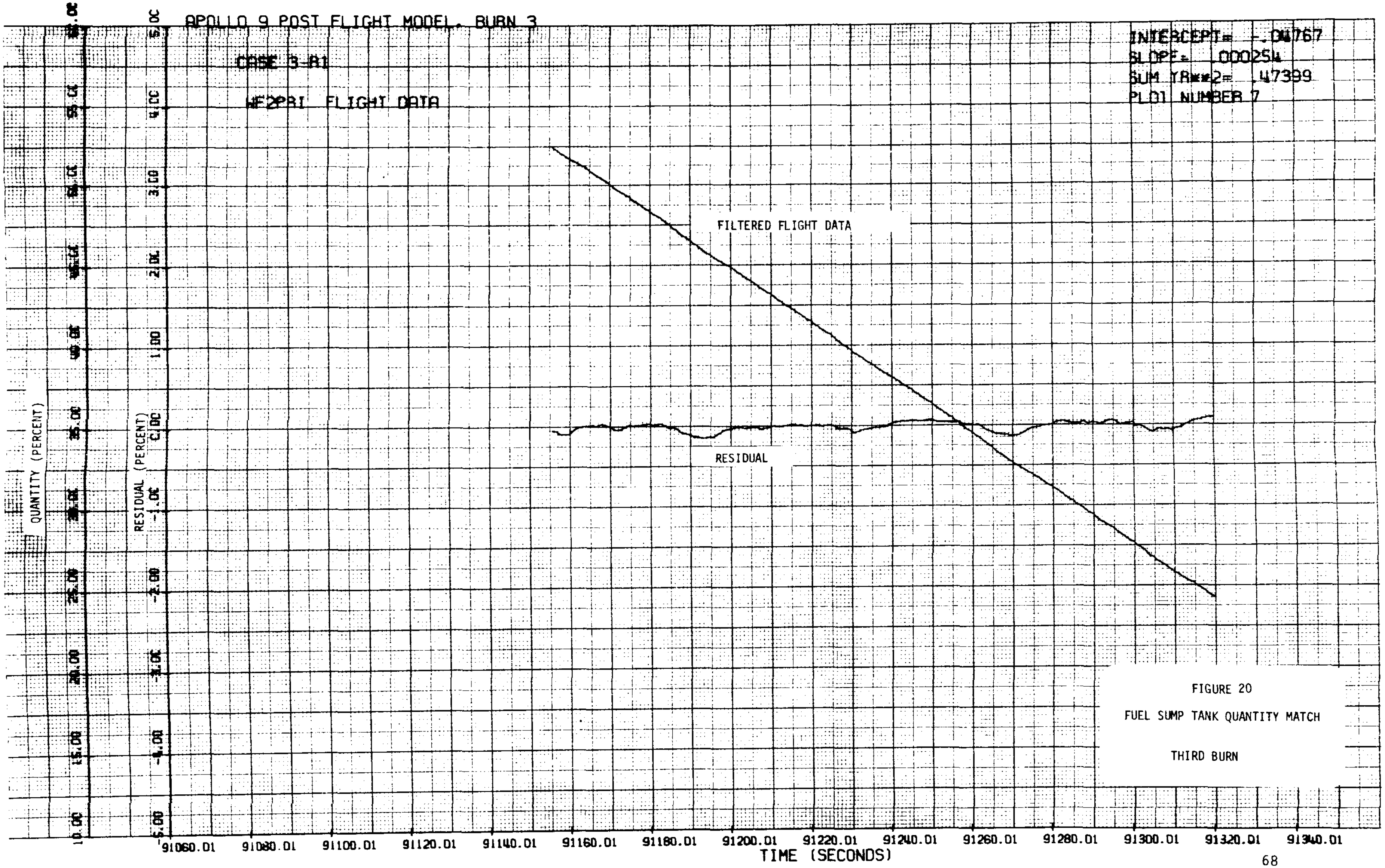


FIGURE 20

FUEL SUMP TANK QUANTITY MATCH

THIRD BURN

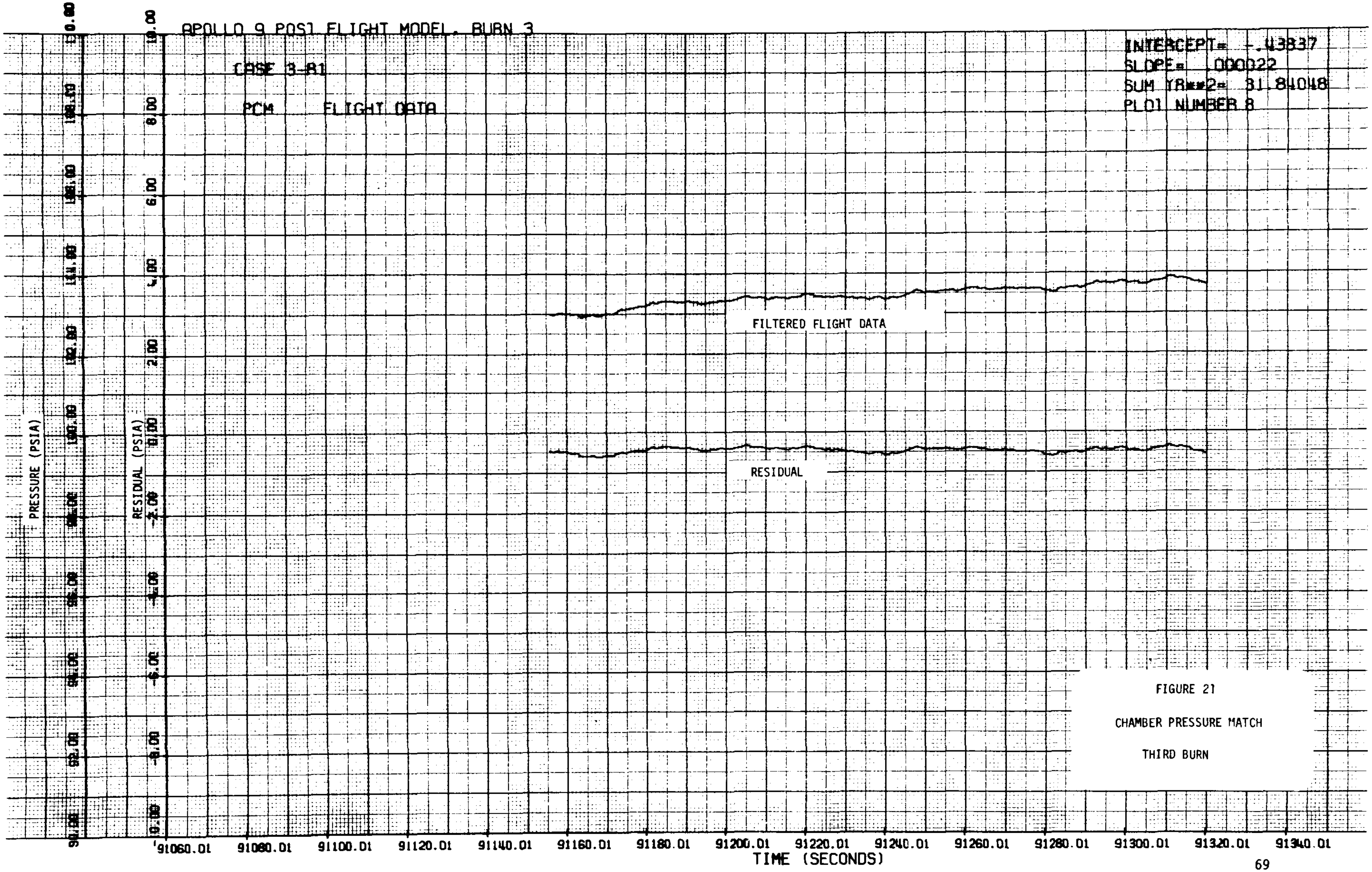


FIGURE 21

CHAMBER PRESSURE MATCH

THIRD BURN

FIGURE 22
THIRD BURN OXIDIZER GAGING SYSTEM
DATA

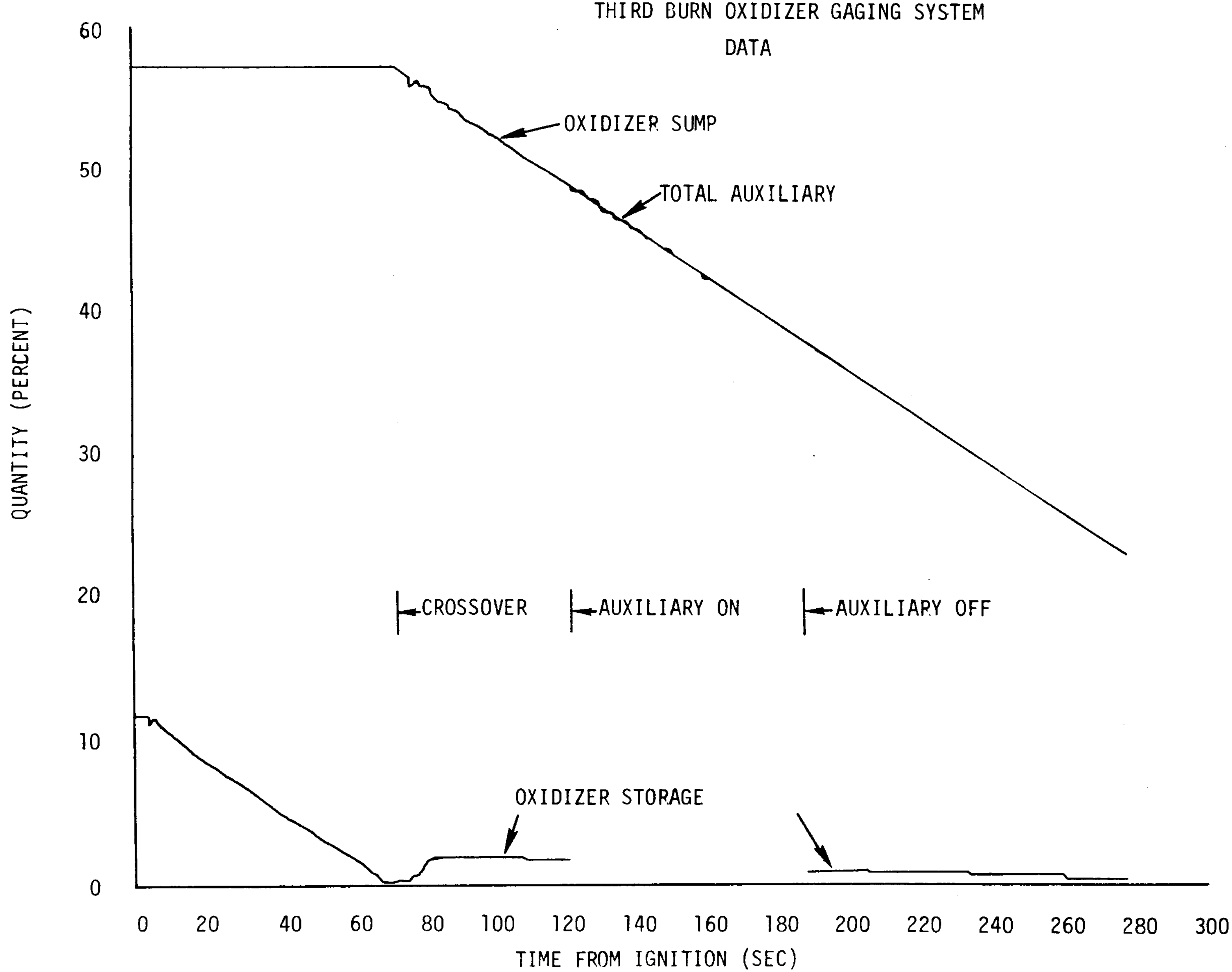


FIGURE 23

THIRD BURN FUEL GAGING SYSTEM DATA

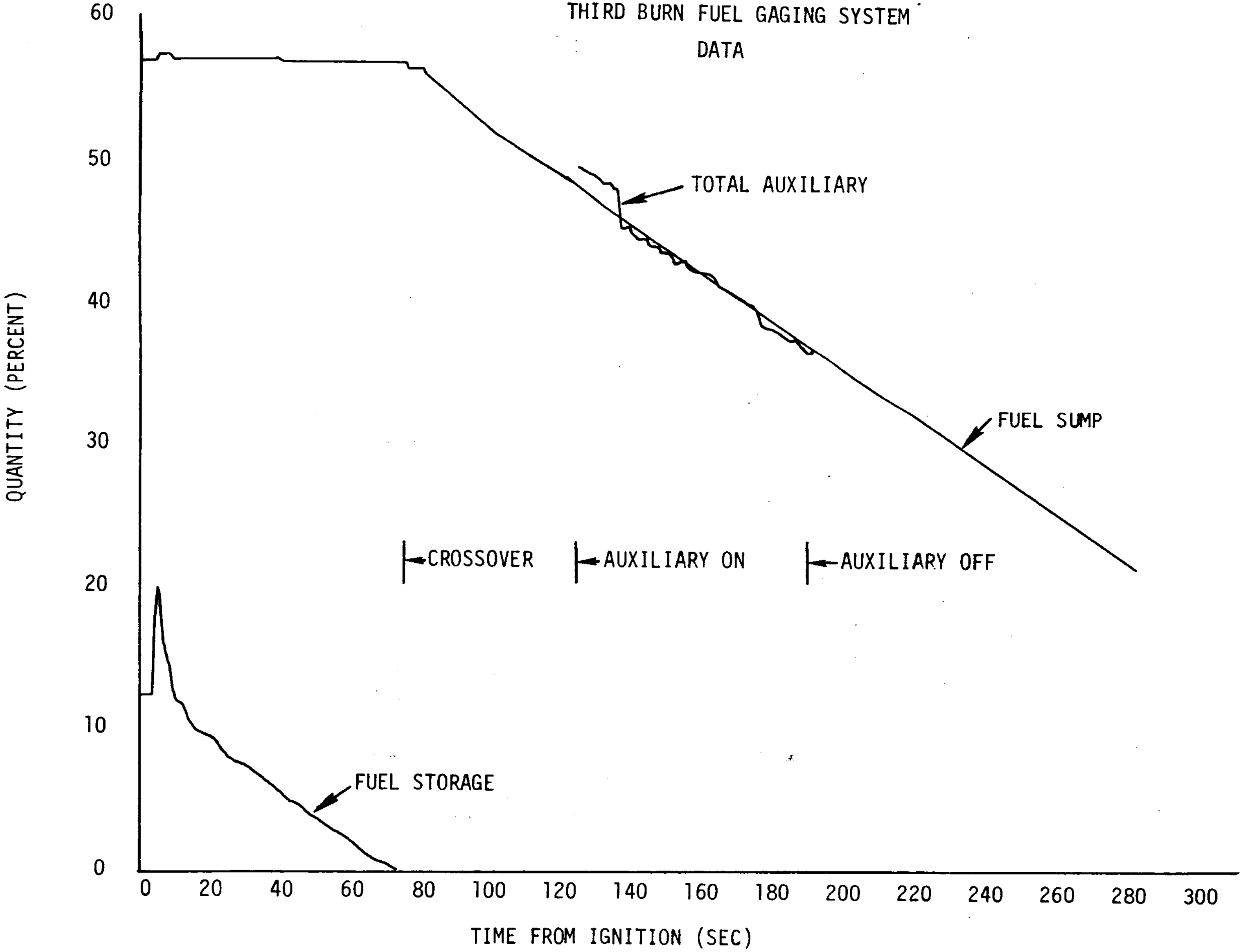


FIGURE 24
INDICATED PROPELLANT UNBALANCE
DURING THIRD BURN